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PRESENTATION OF CLASS I DESIGNS FOR A FAMILY OF COMMUTER AIRPLANES

Prepared for: NASA Grant NGT-~~8001~~ 80001
Prepared by: University of Kansas
AE 790 Design Team
November 1986

Team Leader: Tom Creighton
Team Members: George Dragush
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List of Symbols

Symbol	Definition	Dimension
A	Aspect ratio	-----
b	Wing span	ft
b_a	Aileron span	ft
b_f	Flap span	ft
b_t	Tire width	ft
c	Wing chord	ft
c	Wing mean geometric chord	ft
c_f	Flap chord	ft
c_f	Equivalent skin friction coefficient	-----
c_J	Specific fuel consumption	lbs/lbs/hr
C_D	Drag coefficient	-----
C_{D_o}	Zero lift drag coefficient	-----
c_l	Section lift coefficient	-----
c_{l_α}	Section lift curve slope	1/rad
$c_{l_{\alpha_f}}$	Section lift curve slope with flaps down	1/rad
C_L	Lift Coefficient	-----
C_m	Pitching moment coefficient	-----
D	Drag	lbs
D_p	Propeller diameter	ft
D_t	Tire diameter	ft
d_f, D_f	fuselage diameter	ft
e	Oswald's efficiency factor	-----
E	Endurance	hours
f	Equivalent parasite area	ft2
FAR	Federal Air Regulation	-----
g	Acceleration of gravity	ft/sec2
h	Altitude	ft
i_w	Wing incidence angle	degrees
k_Δ	Sweep angle correction factor	-----
k_f	Correction factor for split flaps	-----
L	Lift	lbs
L/D	Lift-to-drag ratio	-----
l_f	Fuselage length	ft
l_{fc}	Fuselage cone length	ft
l_m	Dist. c.g. to main gear	ft

l_n	Dist. c.g. to nose gear	ft
M	Mach number	-----
n	Load factor	-----
nm	Nautical mile (6,076 ft)	nm
n_p	Number of propeller blades	-----
n_s	Number of struts	-----
N	Number of engines	-----
P	Power, horse-power	hp
P_{bl}	Blade power loading	hp/ft2
q	Dynamic pressure	psf
R	Range	nm
R_n	Reynold's number	-----
RC	Rate of climb	fpm or fps
s	Distance	ft
S	Wing area	ft2
SHP	Shaft horsepower	hp
S_{wet}	Wetted area	ft2
S_{wf}	Flapped wing area	ft2
t	Time	sec, min, hr
t/c	Thickness ratio	-----
T	Thrust	lbs
V	True airspeed	mph, fps, kts
V	Volume coefficient	-----
W	Weight	lbs
X_{ac}	Distance from l.e. c to aerodynamic center	
x, y, z	Distance from reference to a component c.g.	ft, in
x_v, x_h, x_c	Distance from c.g. to a.c. of a surface	ft, in
y_t	Engine-out moment arm	ft

Greek Symbols

α	angle of attack	deg, rad
β	sideslip angle	deg, rad
δ	control surface deflection	deg, rad
λ	taper ratio	-----
Λ	sweep angle	deg, rad
π	3.142	-----
Γ	dihedral angle	deg, rad
ρ	air density	slugs/ft
σ	air density ratio	-----
θ_{fc}	fuselage cone angle	deg, rad
ϕ	lateral ground clearance angle	deg, rad
θ	longitudinal ground clearance angle	deg, rad
θ_{lof}	lift-off angle	deg, rad

ϵ	Downwash angle	-----
ϵ_t	twist angle	deg, rad
η	spanwise station, fraction of the span	-----
ψ	lateral tip-over angle	deg, rad
γ	flight path angle	deg, rad
λ	bypass ratio	-----

Subscripts

a	aileron
A	approach
abs	absolute
cat	catapult
cl	climb
cr	cruise
crew	crew
crit	critical
c/2	semi-chord
c/4	quarterchord
des	design
dry	without fluids or afterburner
e	elevator
E	empty
f	flaps
ff	fuel fraction
F	mission fuel
FL	field length
guess	guessed
h	altitude
h	horizontal tail
le	leading edge
L	landing
LG	landing, ground
LO	lift-off
max	maximum
ME	manufacturer's empty
OE	operating empty
PA	power approach
PL	payload
RC	rate of climb
r	root
res	reserve
reqd	required
s	stall
TO	take-off
TOG	take-off, ground
t	tip
te	trailing edge
tent	tentative
tfo	trapped fuel and oil
used	used
w	wing

wet	wetted
wb	wing-body
wod	wind over the deck

Acronyms

AEO	All engines operating
APU	Auxiliary power unit
B.L.	Buttock line
c.g.	Center of gravity
F.S.	Fuselage station, Front spar
OEI	One engine inoperative
OWE	Operating weight empty
PAX	Passengers
p.d.	Preliminary design
R.S.	Rear Spar
sls	Sea level standard
TBP	Turboprop
W.L.	Waterline

1.0 INTRODUCTION

This report is completed in partial fulfillment of NASA-USRA Grant NGT-8001 requirements. The purpose of this report is to present the class I configuration designs of a family of commuter airplanes.

The proposed commuters range from 25 to 100 passengers. It was decided that all the airplanes in the family should have:

1) 2 aft fuselage mounted engines
2) Low wing
3) T-tail type empennage
4) Tricycle type landing gear

The family concept is introduced in this report in an effort to achieve structural, systems, and handling qualities commonality throughout the passenger range. Implementing commonality can substantially reduce manufacturing and production costs. By achieving common system designs maintenance costs can be reduced by allowing airlines to keep a smaller inventory of spare parts. Therefore, the higher degree of commonality that can be achieved will result in lower direct operating costs and lower life cycle cost. Table 1.1 lists these common features. Attempting to implement many of these commonality requirements has caused configuration design problems. The twin-body concept is introduced in an effort to retain commonality throughout the passenger range.

Chapter 2. discusses the commonality objectives to be designed into the commuter family. Chapter 3. discusses the seven class I configuration designs. Chapter 4. compares the design data to existing airplanes. The extent of structural, systems, and handling qualities commonality achieved will be reviewed in Chapter 5. Conclusions and recommendations are contained in Chapter 6.

TABLE 1.1 COMMON FEATURES DESIRED IN THE ADVANCED TECHNOLOGY COMMUTER FAMILY

FEATURE	IMPLEMENTATION
Fuselage cross section	Completed
Common landing gear Tires and brakes (Both nose and main gear)	Completed
Common landing gear struts and retraction scheme	Completed
Common wing torque box	Completed*
Common empennage torque box	Forthcoming
Common powerplants	Completed**
Common cockpit Instrumentation	Completed
Common flight systems Flight control Fuel Pressurization De-icing	Forthcoming
NLF airfoil technology	Implemented

*Structural analysis in progress

**Two powerplants were selected. A 6000 shp engine, and a 13500 shp engine for the 75 and 100 passenger models.

2. Commonality Objectives for the Commuter Family

The purpose of this chapter is to state the items (structural, systems, operational) that are or will be common to every airplane in the commuter family. After the Class I configurations are presented, an analysis of the extent in which commonality was integrated will be detailed. This is accomplished in Chapter 5.

Commonality of airplanes in the family is an effort to substantially lower acquistion and operating costs for the airplanes. In turn, the airlines will have a wide range of passenger capacity airplanes to operate. A high degree of structural and systems commonality will also result in a smaller spare parts inventory for the airline.

2.1 Fuselage Cross Section

All airplanes in the family have a 4-abreast seating arrangement. The fuselage cross section is presented in Figure 2.1. The rationale for arriving at this decision is given in Appendix A.

2.2 Flight Deck Layout

A preliminary flight deck layout is shown in Figure 2.2. Appendix A describes the flight deck layout and provides a list of cockpit instruments. In the interest of instrument commonality, it was decided that all members of the family have two engines.

2.3 Powerplant Selection

The commuter family utilizes an advanced turbo-prop engine with 10 ft. diameter counter-rotating propellers. From engine sizing requirements discussed in Chapter 3, it was determined that cruise speed and landing fieldlength requirements were critical. These requirements determined the required take-off power for each member of the commuter family.

Two shp models were necessary. A 6000 shp engine powers the 25 to 50 passenger models. A 13,500 shp engine powers the 75 and 100 passenger models. For some of the airplanes, it is necessary to derate the engine horsepower. Table 2.1 presents required take-off power requirements and derated horsepowers for the commuter family.

Derating some of the engines will allow for longer service life because engine cores will not have to burn as hot and will be able to last longer. Figure 2.3 presents dimensioned view of the PD436-11 powerplant. The engines used in the commmuter family are scaled from this engine.

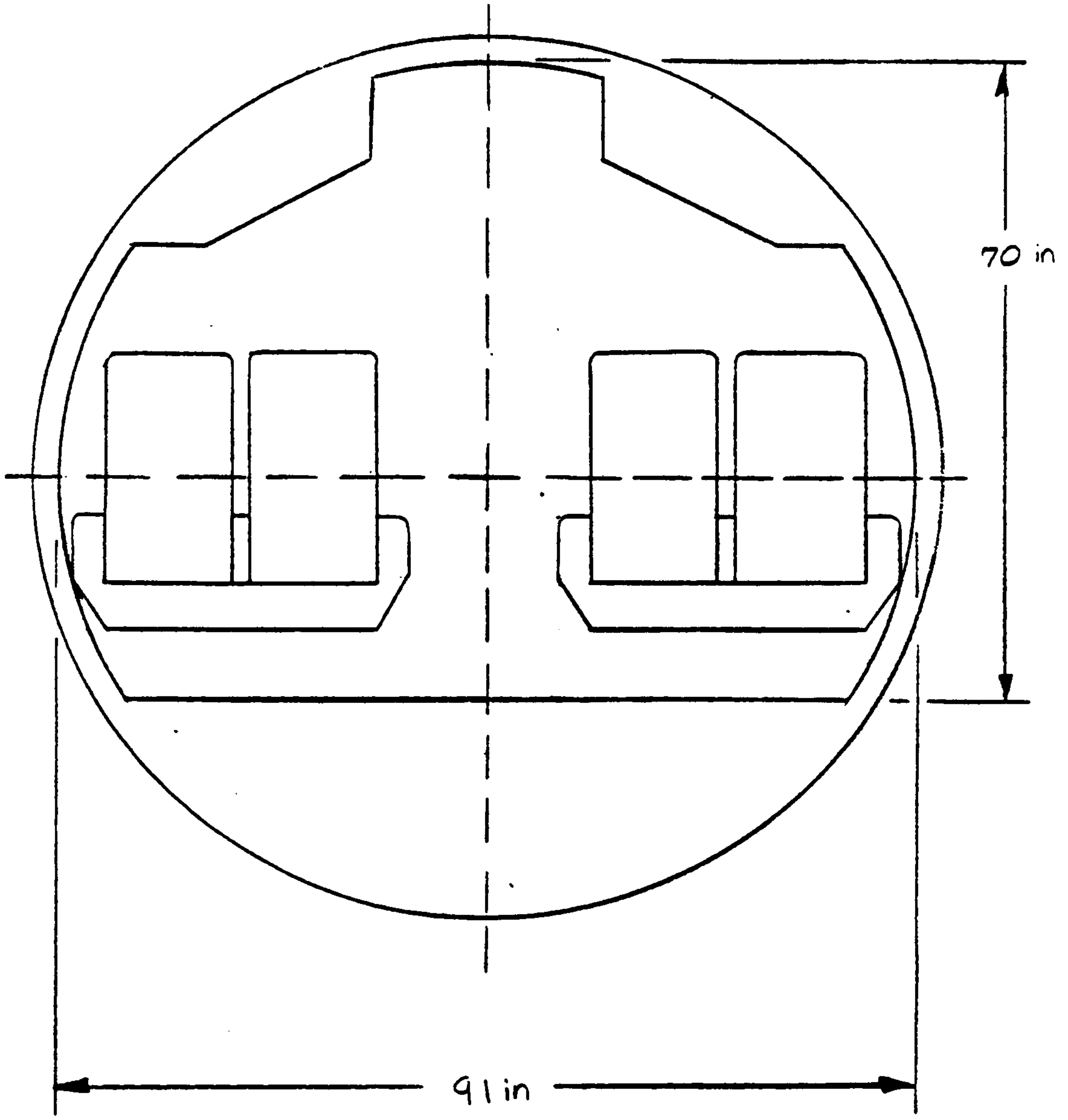

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FIGURE 2.1 FUSELAGE CROSS SECTION

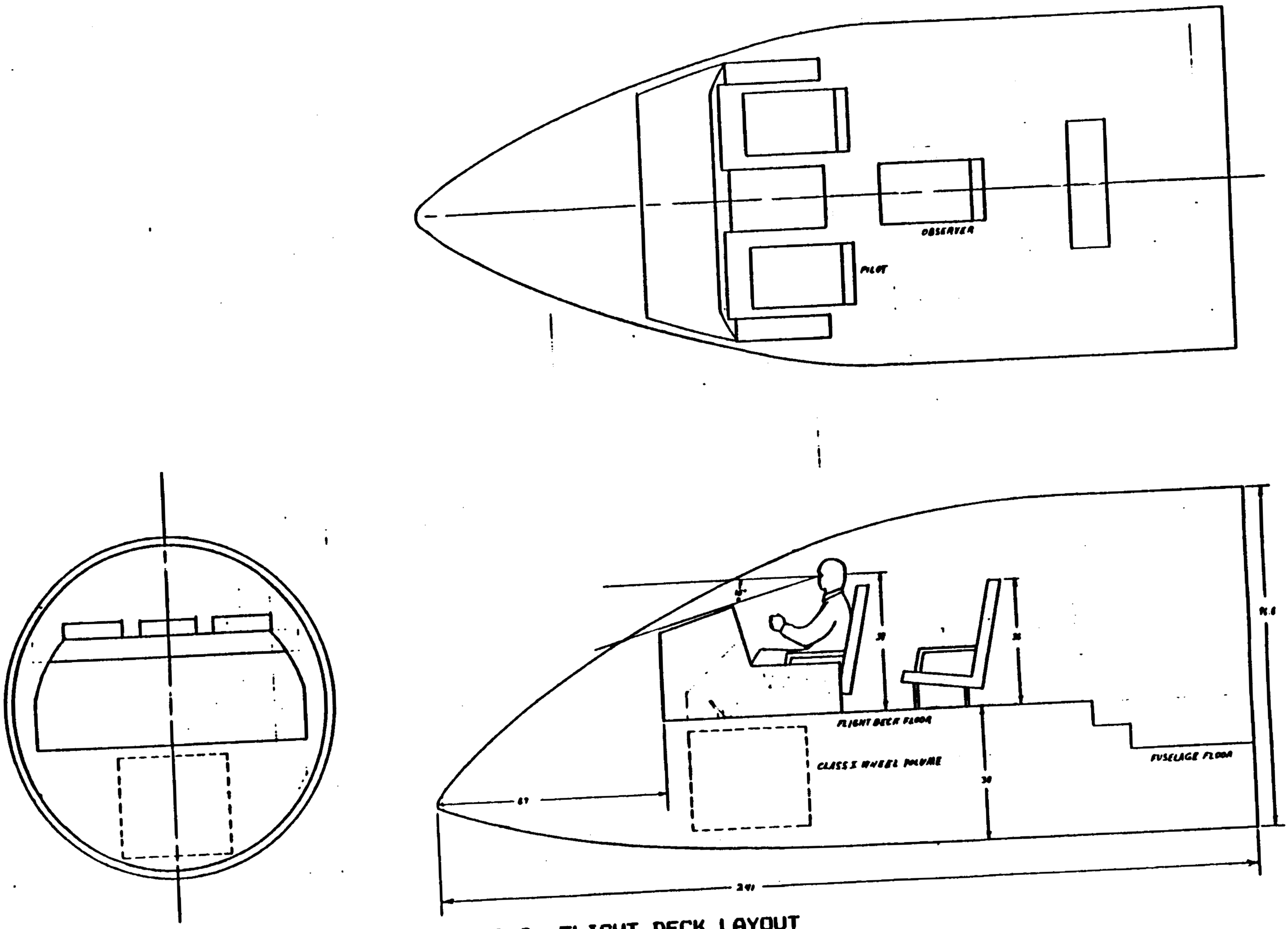

FIGURE 2.2 FLIGHT DECK LAYOUT

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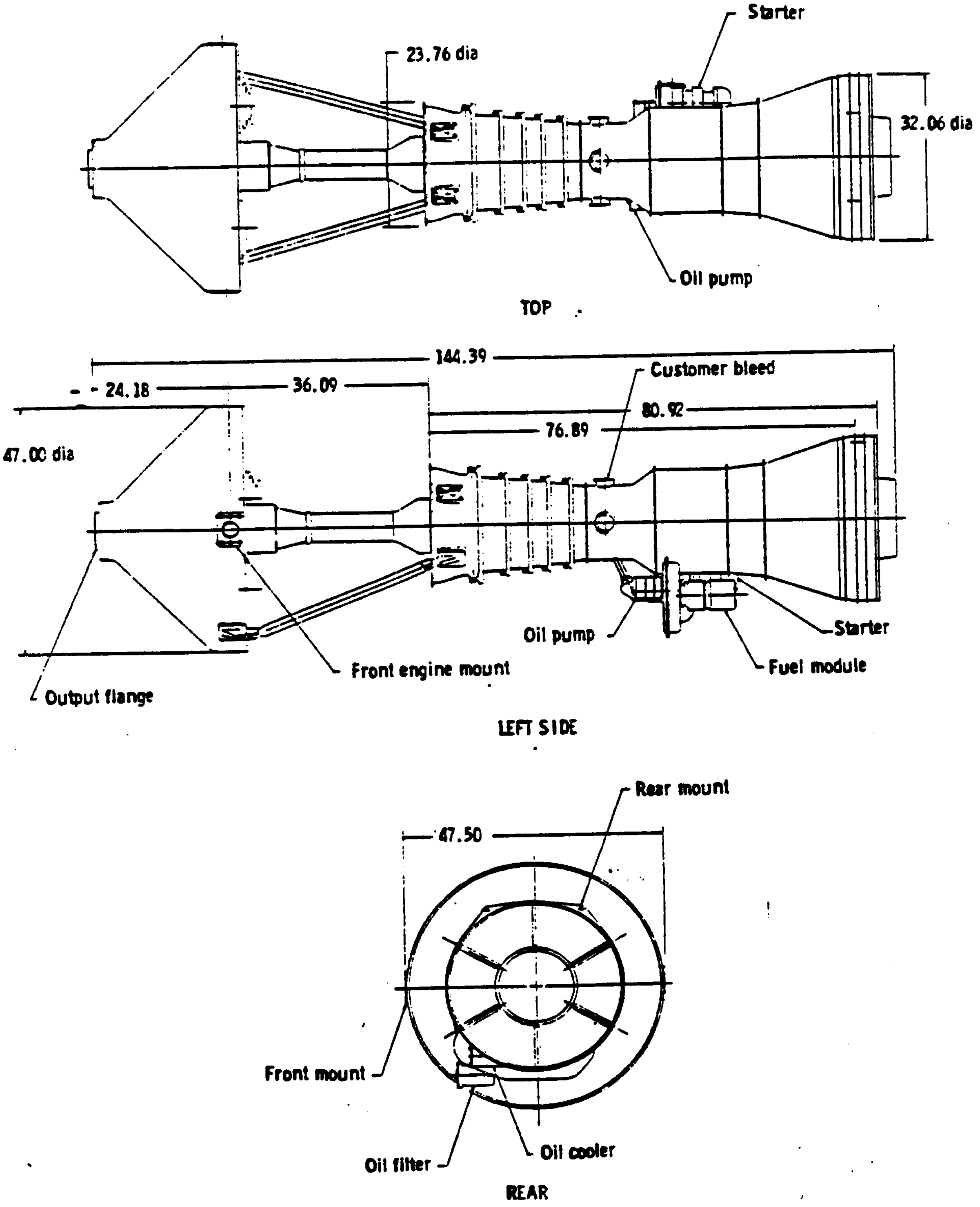

Note: All dimensions are in inches

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Figure 2.3 PD436-11 Powerplant

Table 2.1--Engine Power Requirements.

Passenger Model	Take-Off Power (shp)	Total Engine Power (shp)	Derated Engine Power (shp)
25 Passenger	8,419	2x6000	2x4500
36 Passenger	8,970	2x6000	2x4500
50 Passenger	11,000	2x6000	------
75 Pass. (conv.)	19,640	2x13,500	2x10,000
100 Pass. (conv.)	26,750	2x13,500	------
Twin-body 75 Pass.	18,000	2x13,500	2x9,000
Twin-body 100 Pass.	22,000	2x13,500	2x11,000

2.4 Wing and Airfoil Design

A natural laminar flow airfoil similiar to the HSNLF(1)-0213 is used on all members of the commmuter family. Appendix C presents the airfoil cross section and design data. Table 2.2 contains Reynolds numbers for the wings. Transition Reynolds numbers directly related to the amount of laminar flow obtained on the airfoil. These Reynolds numbers range from approximately 11 to 30 million. As the Reynolds number increases over the wing, less chordwise laminar flow is realized.

To minimize induced drag an aspect ratio 12 cantilever wing was designed for all airplanes in the commuter family. The high aspect ratio translates into a relatively heavy wing. Appendix O contains a wing weight trade study. Table 2.3 contains the wing planform geometry for all of the commuter family.

2.5 Landing Gear

All landing gear, nose and main, have the same 30" x 9" tire. The main gear wheel base and retraction scheme is desired to be the same. This allows for similar strut sizing for the airplanes. Appendix D contains the main gear retraction scheme for the commuter familuy. A landing gear tire size study is also included in Appendix D. Table 2.4 provides the number and size of the tires on each gear strut.

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Table 2.2--Wing Reynolds Numbers for the Commuter Family.

Passenger Model	$R_{N_{root}}$ $(\times 10^6)$	$R_{N_{tip}}$ $(\times 10^6)$
25 Pax	16.9	6.8
36 Pax	17.4	7.0
50 Pax	19.9	8.0
75 Pax (conv.)	28.2	11.3
100 Pax (conv.)	32.9	13.2
Twin-body 75 Pax	17.4	7.0
Twin-body 100 Pax	19.9	8.0

Table 2.3--Wing Geometry of the Commuter Family.

Passenger Model Parameters	25 Pax	36 Pax	50 Pax	75 Pax conv	100 Pax conv	75 Pax twin	100 Pax twin
Area, S (ft^2)	421	449	591	1178	1604	722	923
Span, b (ft)	71.1	73.4	84.3	119	139	105	118
Aspect ratio, A	12.0	12.0	12.0	12.0	12.0	15.1	15.1
MGC, $\bar{c}$ (ft)	6.28	6.50	7.46	10.5	11.6	7.50	8.33
Taper ratio,	0.4	0.4	0.4	0.4	0.4	0.4	0.4
Leading edge sweep, (deg)	15	15	15	15	15	15	15
Dihedral, (deg)	7	7	7	7	7	7	7
Thickness, t/c	.13	.13	.13	.13	.13	.13	.13

2.6 Wing Torque Box

Figure 2.4 presents the 25, 36, and 50 passenger wing planforms with the torque boxes included. These wing planforms are also utilized on the twin body concepts presented in Chapter 3. A common wing carry thru structure is possible if these three planforms are used throughout the family.

Figure 2.5 presents the wing cross sections. The torque box structure is common to all the wing sections. The L.E. and T.E. sections are faired in to retain as much of the NLF airfoil characteristics as possible. Appendix G contains the design work computed for this proposal.

2.7 Tailcone Arrangements

All airplanes in the family have the same fuselage tailcone on all the airplanes. It is desired to keep the vertical tail root spar locations identical positions on all tailcones. When Class II weight and balance work is concluded, a common empennage arrangement will be proposed. Table 2.5 contains empennage geometric data for the commuter family.

2.8 Systems Commonality

Common system design will be attempted for the following systems:

1. Fuel system.
2. Flight controls.
3. Hydraulics.
4. Pressurization.
5. De-icing.

2.8.1 Fuel System

All airplanes in the commuter family carry fuel in the wing. Since a common wing torque box arrangement is proposed, some of the integral fuel tanks can possibly be the same on all airplanes. However, the varying wing spans and required fuel volumes will not allow for complete system commonality. Similar vents and access panels will be incorporated into all members of the family. Fuel flow rates will determine if similar fuel pumps can be used on all family members.

2.8.2 Flight Control System

A separate surface stability augmentation system is proposed to achieve identical handling qualities throughout

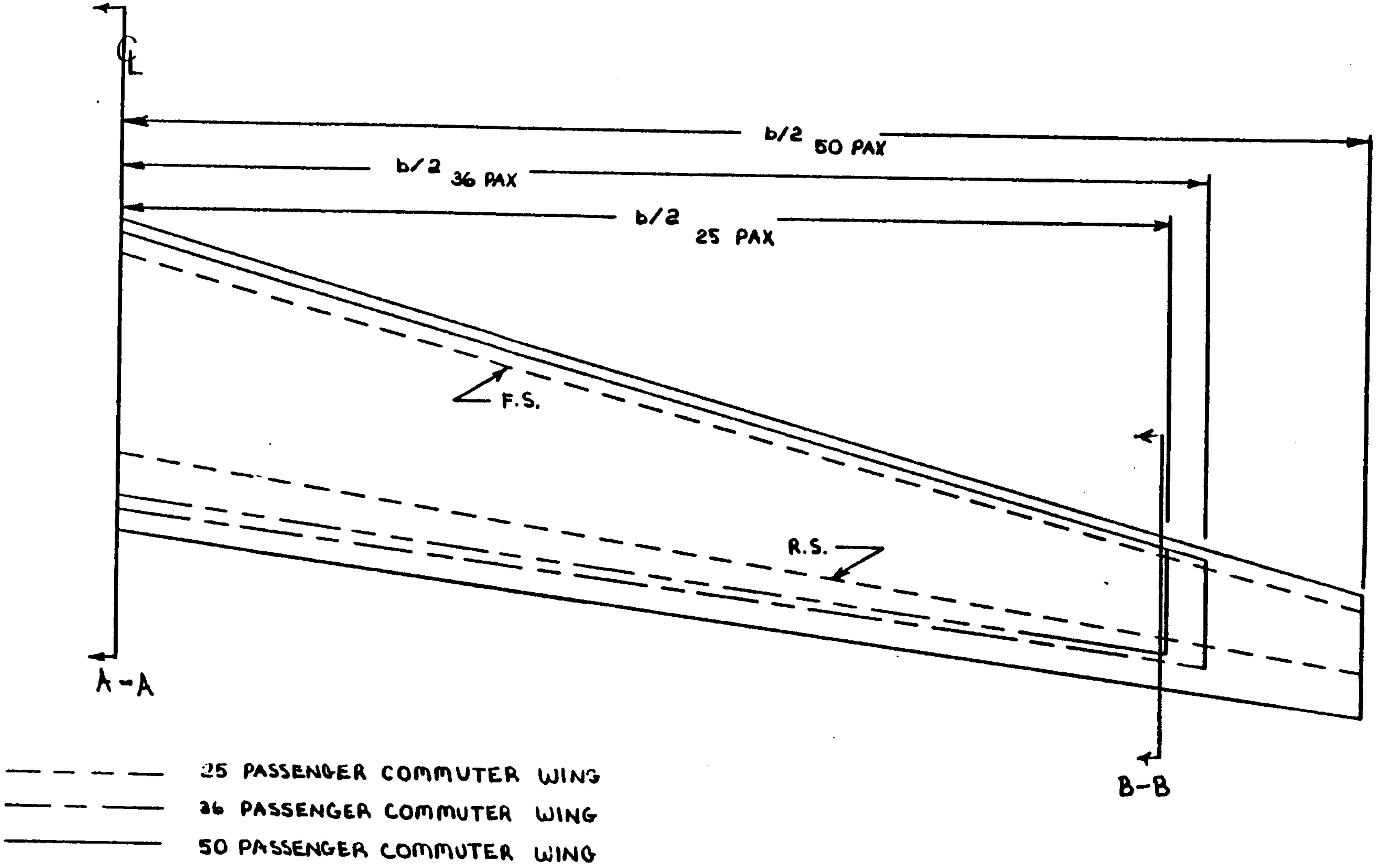
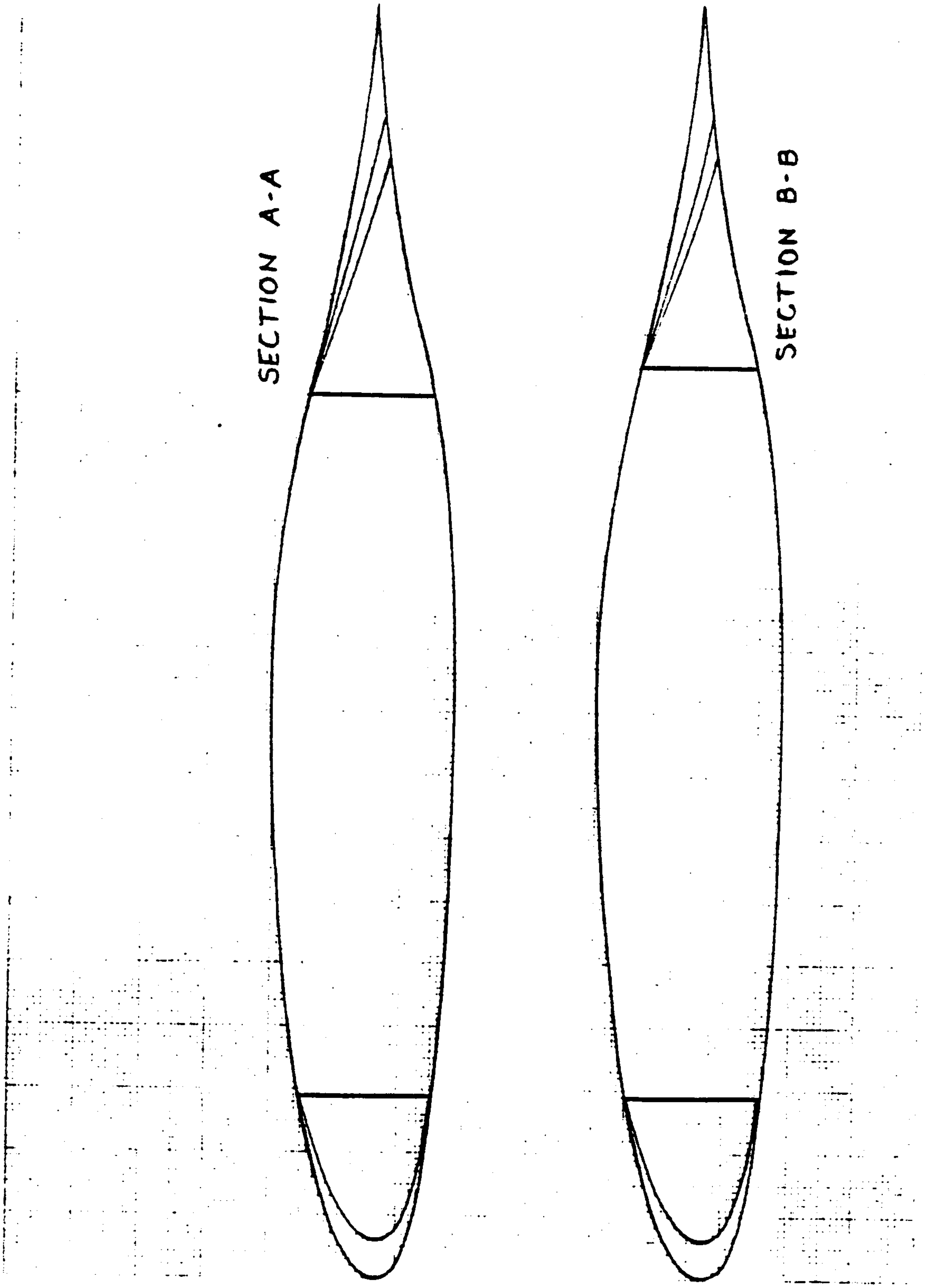


FIGURE 2.4 WING TORQUE BOXES



CALC	G. Drazach	11/21/86	REVISED	DATE	
CHECK					Figure 2.5 Wing Cross Sections
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Table 2.4--Landing Gear Tire Sizes.

Passenger Model	Nose gear	Main gear (per strut)
25 Pax	2 x 30" x 9"	2 x 30" x 9"
36 Pax	2 x 30" x 9"	2 x 30" x 9"
50 Pax	2 x 30" x 9"	2 x 30" x 9"
Conv. 75 Pax	2 x 30" x 9"	2 x 30" x 9"
Conv. 100 Pax	2 x 30" x 9"	4 x 30" x 9"
Twin 75 Pax	2 x 30" x 9"	2 x 30" x 9"
Twin 100 Pax	2 x 30" x 9"	2 x 30" x 9"

Table 2.5--Empennage Geometry for the Commuter Family.

Passenger Model Parameters	25 Pax	36 Pax	50 Pax	75 Pax conv.	100 Pax conv.	75 Pax twin	100 Pax twin
Horizontal Tail:							
Area, S_H (ft^2)	69	69	102	134	155	102	102
Span, b_H (ft)	16.6	16.6	22.6	26.7	28.7	22.6	22.6
MGC, $\bar{c}_H$ (ft)	4.20	4.20	4.68	5.42	5.40	4.68	4.68
Aspect ratio, A	4.0	4.0	5.0	5.3	5.3	5.0	5.0
Taper ratio, λ	0.7	0.7	0.5	0.35	0.35	0.5	0.5
L.E. sweep, (deg)	20	20	25	22	25	25	25
Vertical tail:							
Area, S_V (ft^2)	170	130	170	363	303	130	140
Span, b_V (ft)	14.0	12.0	15.4	22.5	20.6	12.0	15.4
MGC, $\bar{c}_V$ (ft)	13.3	11.9	11.4	16.4	15.0	11.9	9.40
Aspect ratio, A	1.15	1.10	1.40	1.40	1.40	1.10	1.70
Taper ratio, λ	0.3	0.3	0.5	0.6	0.6	0.3	0.5
L.E. sweep, (deg)	54	58	40	42	45	58	40

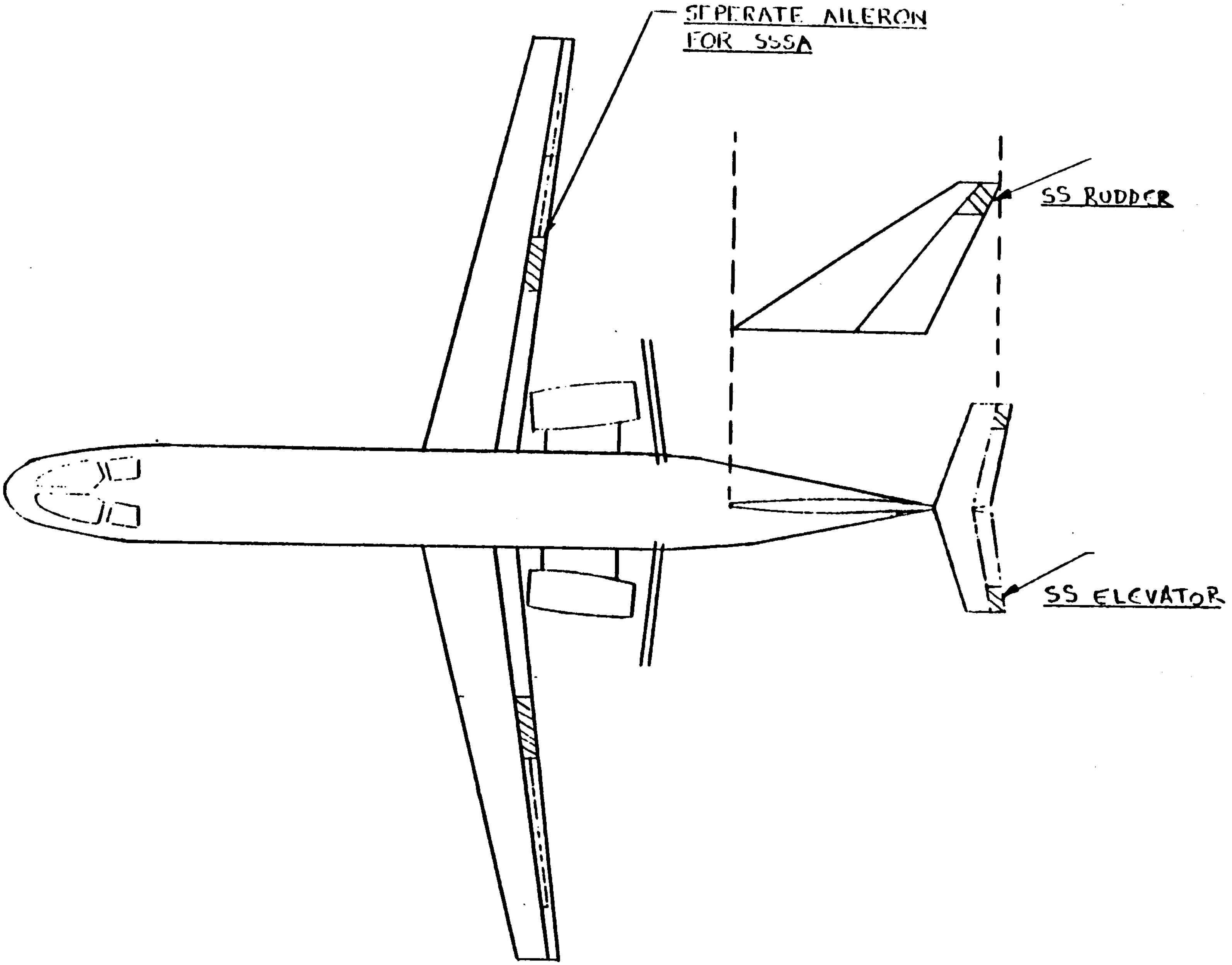

FIGURE 2.6 EXAMPLE OF A PROPOSED SSSA FLIGHT CONTROL SYSTEM

the passenger range. This system will make use of electro-hydrostatic actuation. A particualar actuator has not yet been decided upon. A control system design has not yet been completed. Figure 2.6 shows a proposed separate surface stability augmentation system that could be incorporated into the commuters.

2.8.3 Hydraulic System

A common operating pressure hydraulic system will be implemented for the landing gear actuation. Further study is neccesary to determine the operating capabilities of this system.

2.8.4 Pressurization System

All passenger cabins in the family are pressurized to a 5000 ft. atmosphere at 30,000 ft. All airplanes willutilize the same pressurization system.

2.8.5 De-Icing System

The T.K.S. de-icing system, which will also double as a bug-cleaner, will be implemented into the commuter family. The T.K.S. system is a liquid ice protection system that distributes a solution onto the leading edge of the wing through a porous wing skin. Cleaning the leading edge is required to preserve the laminar flow over the wing. The L.E. volume of the wings will be checked to see if one size system can be implemented.

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3. PRESENTATION OF CLASS I DESIGNS

The purpose of this chapter is to document seven class I configurations for the Advanced Technology Commuter Family. The reason for developing these baseline designs is to have a series of reasonably firm configurations on which to perform realistic studies of the feasibility of achieving the commonality goals stated in Table 1.1. The baseline designs evolved from a set of mission specifications listed in Table 3.1

Sections 3.1 through 3.7 address the class I design evolution of these baseline designs.

Section 3.1 presents the 25 passenger model, the smallest capacity airplane in the family. The subject of section 3.2 is the 36 passenger derivative. The 36 passenger configuration was used to develop a 75 passenger twin fuselage configuration. This 75 passenger configuration is the subject of section 3.3. Section 3.4 presents the 50 passenger derivative. Section 3.5 presents a 100 passenger twin fuselage design. This twin-fuselage was developed from the 50 passenger model. Section 3.6 and 3.7 presents 75 and 100 passenger derivatives that are of conventional configuration. It was found that implementation of many commonality objectives were not possible with these large conventional configurations. A commonality analysis is the subject of chapter 5.

TABLE 3.1 Mission Specification for the Commuter Family

	25 pax	36 pax	50 pax	75 pax	100 pax
Payload (lbs)	5125	7380	10250	15375	20500
Crew (lbs)	410	615	615	820	820
Range (n.m.)	1100	1100	1100	1500	1500
Altitude	All Cruise at 30,000 ft.				
Cruise Speed	All Cruise at Mach .70				
Climb	All Climb-out at 3000 fpm				
TOFL, LFL	All Field Lengths are 3,500 ft				
Powerplants (shp)	6000	6000	6000	13500	13500
Derated (shp)	4500	4500	6000	9000	13500
Pressurization	All Pressurized 5000 ft at 30000 ft				
Certification	All FAR 25				

3.1 PRELIMINARY DESIGN OF THE 25 PASSENGER BASELINE CONFIGURATION

Figure 3.1.1 contains the class I 3-view for the 25 passenger commuter. Table 3.1.1 contains the geometry of the configurations

3.1.1 INITIAL WEIGHT AND PERFORMANCE SIZING FOR THE 25 PASSENGER BASELINE CONFIGURATION

3.1.1.1 INITIAL WEIGHT SIZING

Initial weight sizing was conducted using a method in Reference 1. The following assumptions were made for the airplane:

1) $(L/D)_{cr} = 16$
2) $C_p = 0.4$ lbs/hp/hr

The above assumptions and the mission specifications, given in Table 3.1.2, yielded the airplane weights and sensitivities in Table 3.1.3. Appendix H, section H.2 contains output from XEWTOG, a computerized weight sizing method developed at the University of Kansas.

3.1.1.2 INITIAL PERFORMANCE SIZING

XPRFRM, a computer program developed at the University of Kansas, was used to determine the required take-off power, P_{TO} and wing area, S that meet the performance criteria given in Table 3.1.2. XPRFRM follows the method of Reference 1. Maximum lift coefficients and wing aspect ratio are also determined. Figure 3.1.2 shows the required power loading, wing loading combinations that satisfy the performance criteria. From Figure 3.1.2 it is determined that cruise speed and landing field length requirements are critical for this airplane. The results of the performance sizing effort are listed in Table 3.1.2. Appendix H, section H.3 details the computer output of XPRFRM.

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TABLE 3.1.1 TABLE OF GEOMETRY FOR THE 25 PASSENGER COMMUTER

	WING	HORIZONTAL TAIL	VERTICAL TAIL
S ft^2	421	69	170
b ft	71.1	16.6	14
$\bar{c}$ ft	6.28	4.2	13.33
$\bar{c}$ LE F.S.	487 in	962 in	795 in
A	12	4	1.15
Λ_{LE}	15°	20°	54°
λ	.4	.7	.3
t/c	.13 root	.11	.11
Airfoil	NLF	NLF (sym)	NLF (sym)
Γ	7°	0°	0°
i	0°	0°	0°
		elevator chord ratio .36	rudder chord ratio .35

Spoiler: chord ratio .08
span ratio .50 to .90

Flap: chord ratio .15
span ratio .11 to 1.0

	FUSELAGE	CABIN INTERIOR	OVERALL
Length ft	69.4	22.9	74.6
Height in	96	76	320
Width in	96	91	852

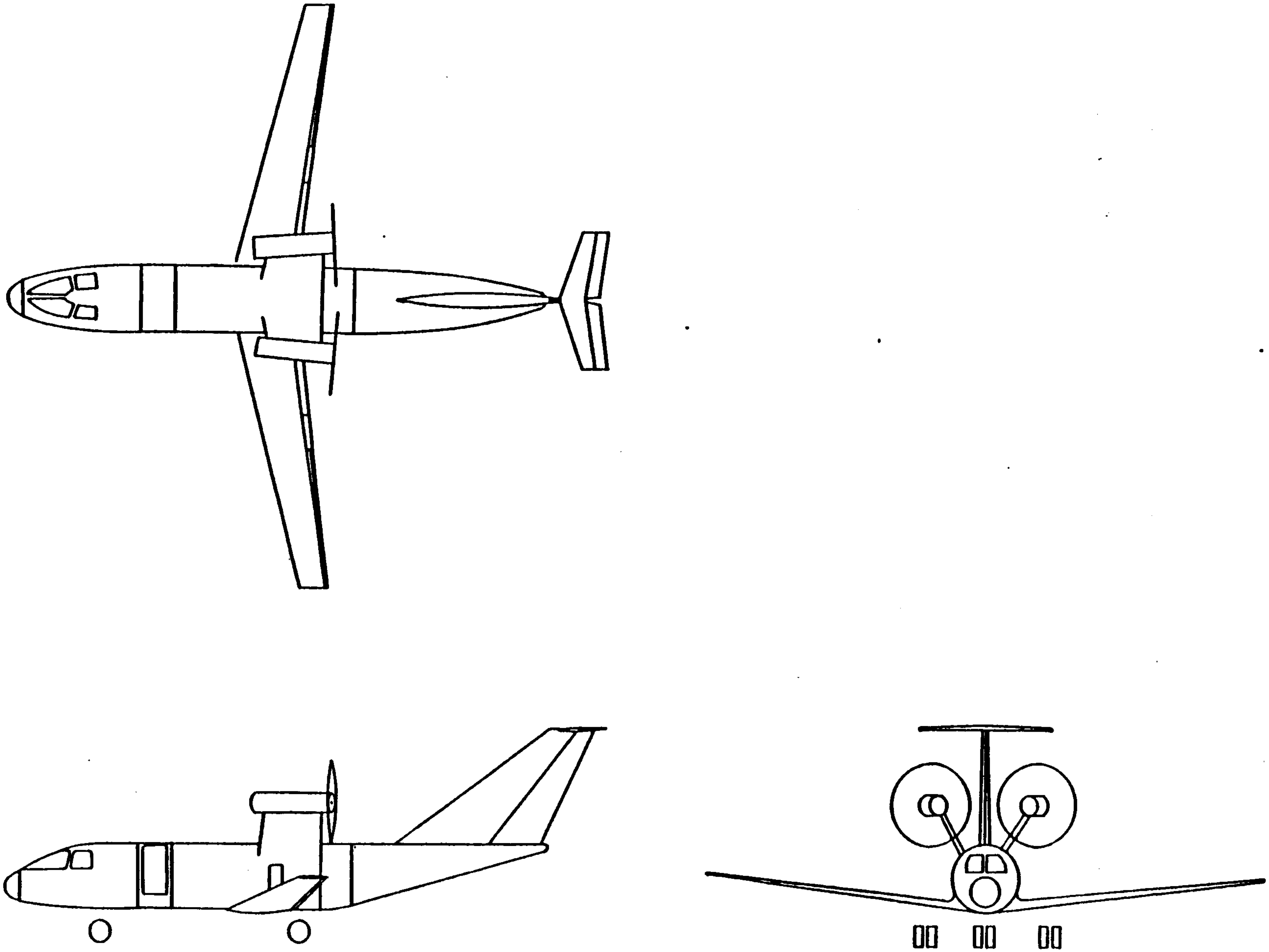

FIGURE 3.1.1 3-VIEW OF THE 25 PASSENGER MODEL

TABLE 3.1.2 MISSION SPECIFICATION FOR A 25 PASSENGER ADVANCED TECHNOLOGY COMMUTER AIRPLANE

PAYLOAD:	25 passengers at 175 lbs each with 30 lbs of baggage per passenger, carry-on luggage capability is required
CREW:	2 pilots at 175 lbs each with 30 lbs of baggage each
RANGE:	1100 nm with maximum payload with 25% fuel reserves
ALTITUDE:	30,000 ft at the design range
CRUISE SPEED:	Mach = 0.70
CLIMB:	climb rate of 3000 fpm
TAKE-OFF AND LANDING:	3500 ft balanced field length
POWERPLANTS:	advanced turboprops
PRESSURIZATION:	5000 ft cabin at 30,000 ft
CERTIFICATION BASE:	FAR 25

MISSION PROFILE:

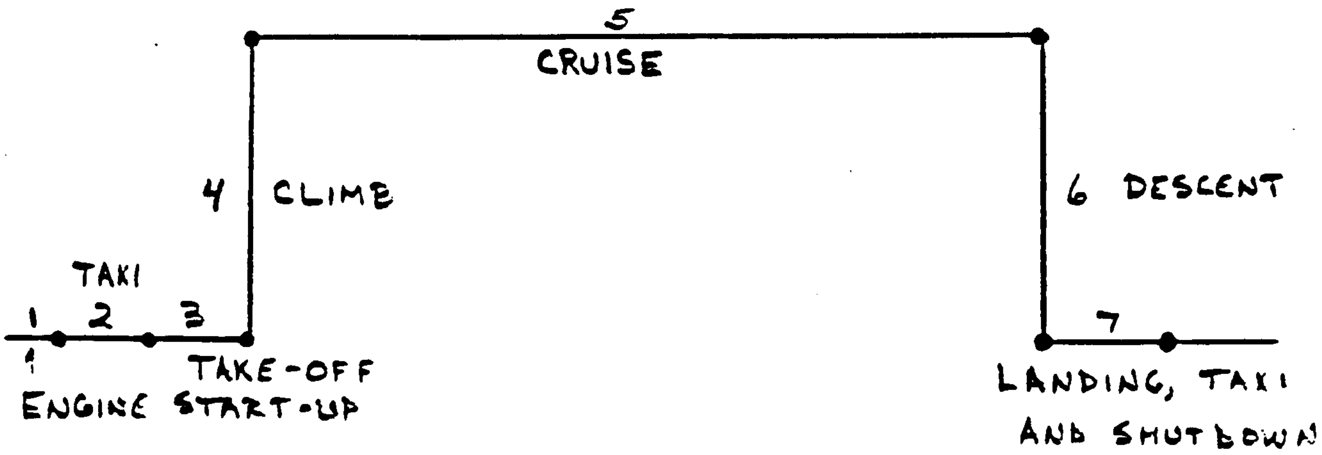

TABLE 3.1.3 INITIAL SIZING PARAMETERS FOR THE 25 PASSENGER COMMUTER

Weights: Take-off Weight - W_{TO} = 21046 lbs
Operating Weight Empty - W_{OE} = 12154 lbs
Payload Weight - W_{PL} = 5125 lbs
Crew Weight - W_{CREW} = 410 lbs
Mission Fuel Weight - W_F = 3767 lbs

Wing Area - $S = 421\ ft^2$
Wing Aspect Ratio - $A = 12$
Take-off Power - $P_{TO} = 8419$ shp

Required Lift Coefficients -
Clean $C_{L_{MAX}} = 1.4$
Take-off $C_{L_{MAX}} = 1.4$
Landing $C_{L_{MAX}} = 2.2$

Take-off Weight Sensitivities -

$$\frac{\partial W_{TO}}{\partial C_p} = 20026.3 \quad (lb/lb/hp/hr)$$

$$\frac{\partial W_{TO}}{\partial \eta_p} = -9424.2 \quad (lbs)$$

$$\frac{\partial W_{TO}}{\partial (L/D)} = -500.7 \quad (lbs)$$

$$\frac{\partial W_{TO}}{\partial R} = 7.3 \quad (lb/nm)$$

3.1.2 FUSELAGE AND COCKPIT LAYOUTS

The 25 passenger airplane has the same flight deck layout and fuselage cross section as the rest of the commuter family. The cockpit design and the fuselage cross section are contained in Appendix A. The lengths of the fuselage and cabin are given in Table 3.1.1.

The design methodology followed the steps in Reference 2. and 3.

3.1.3 ENGINE SELECTION

The commuter family will be powered by 2 advanced turboprop engines. The 25 passenger requires the use of two 6000 shp turboprops.

Appendix B contains engine data for the airplane.

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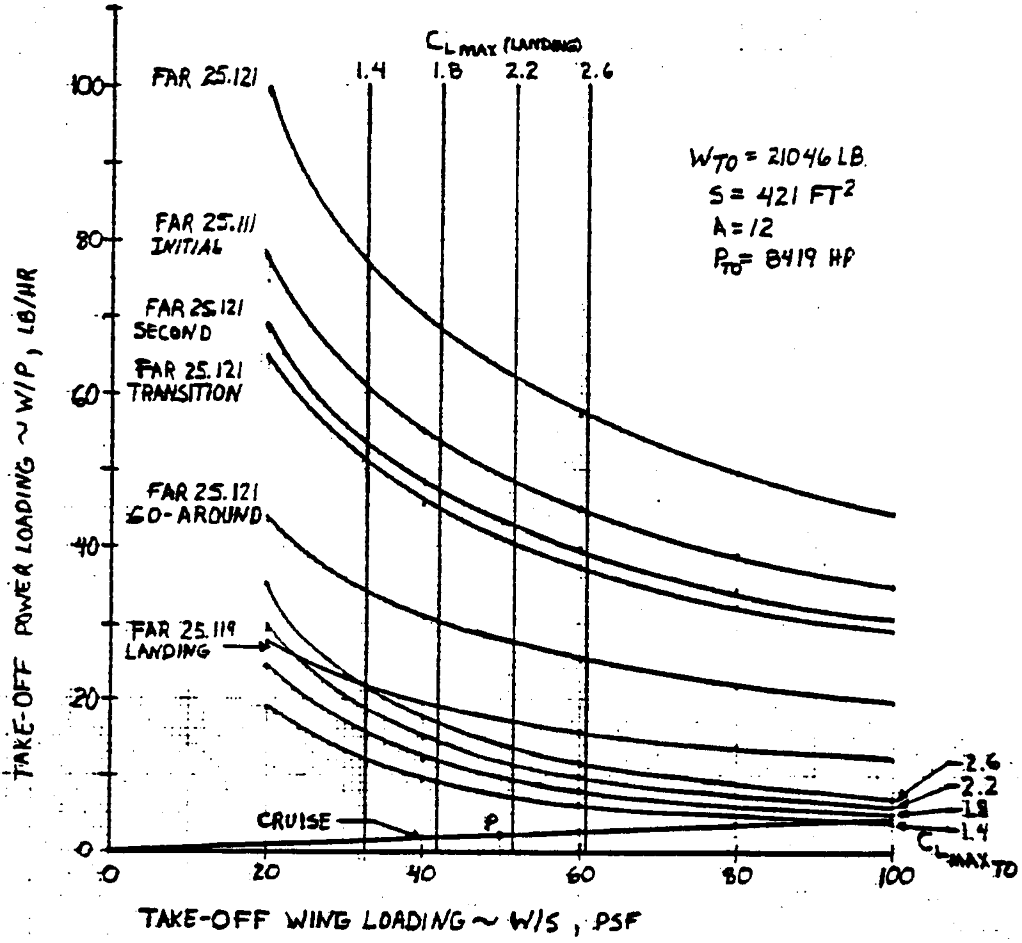

CALC	D. HENSLEY	11-17-86	REVISED	DATE	**FIGURE 3.1.2** **PERFORMANCE MATCHING OF** **THE 25 PASSENGER MODEL**
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3.1.4 WING AND FLAP DESIGN

Table 3.1.1 presents the geometry of the wing and flaps. Parameters such as leading edge sweep and wing thickness were dictated by the selection of an NLF Airfoil. Appendix C contains the airfoil cross section and airfoil parameters. Wing parameters were selected using the method of Reference 2. chapter 6.

The flaps were sized to a $C_{L_{MAX_L}} = 2.2$. This required the use of fowler flaps. The sizing methods used are contained in chapter 7 of Reference 2. The design calculations are in Appendix H, section H.4.

3.1.5 DESIGN OF THE EMPENNAGE

Table 3.1.1 shows the empennage for the 25 passenger airplane. Initially the V-bar method of chapter 8 in Reference 2. was used to size the empennage. The design calculations are in Appendix H, section H.5. The initial tail areas that resulted are listed below:

$$S_H = 51 \text{ ft}^2$$

$$S_V = 57 \text{ ft}^2$$

The empennage was redesigned from stability and control considerations. These considerations are discussed in section 3.1.9.

3.1.6 CONTROL SURFACE SIZING

3.1.6.1 LATERAL - DIRECTIONAL CONTROLS

Since full span flaps were required for landing, spoilers were used in place of ailerons. The spoiler geometry was determined from chapter 8 of Reference 2. Spoiler geometry is contained in Table 3.1.1. The rudder was also sized from methods in chapter 8 of Reference 2. Its geometry is contained in Table 3.1.1.

3.1.6.2 LONGITUDINAL CONTROLS

The elevators were sized using methods in chapter 8 of Reference 2. The geometry of the elevator is contained in Table 3.1.1

3.1.7 LANDING GEAR DESIGN

From Reference 2. chapter 9. it was determined that a 30" x 9" tire could be utilized for the nose and main landing gear on every airplane of the commuter family. A preliminary retraction scheme for the main gear is shown in Appendix D. The gear placement was dictated by the weight and balance

calculations shown in section 3.1.8. Lateral tip-over, and longitudinal gear placement criteria given in Reference 2. were met. Appendix H, section H.6 contains the lateral tip-over calculations.

3.1.8 CLASS I WEIGHT AND BALANCE CALCULATIONS

Class I component weights were calculated by averaging typical take-off weight fractions of commuter airplanes. Appendix F contains the class I weight fractions for the commuter family. Using methods in chapter 10 of Reference 2. A preliminary weight and balance of the 25 passenger commuter was determined. Component weights and center of gravity locations are contained in Table 3.1.4. A general arrangement drawing is contained in Figure 3.1.3. The center of gravity excursion diagram is contained in Figure 3.1.4. The 25 passenger commuter has a 13.4" excursion range. This is .18 $\bar{c}_w$.

3.1.9 STABILITY AND CONTROL RESULTS

A class I stability and control analysis was performed using the methods of Reference 2. chapter 11. Table 3.1.5 contains geometric quantities and stability derivatives necessary to size the empennage from stability and control considerations. Design calculations are located in Appendix H, section H.7.

3.1.9.1 LONGITUDINAL STABILITY

From methods in chapter 11. of Reference 2. the horizontal tail was resized to incorporate a desired static margin of 5%. Appendix H, Figure H.2 presents the longitudinal X-plot for the airplane. From this plot it is seen that a tail area of 66 ft^2 is required. Because this required horizontal tail area is very similar to that required for the 36 passenger configuration, it was decided to implement the tail required for the 36 passenger airplane on both configurations. This is a very acceptable compromise between performance requirements and commonality.

3.1.9.2 LATERAL - DIRECTIONAL STABILITY

From methods in chapter 11 of Reference 2. the vertical tail area required to hold engine-out flight was determined to be critical. Appendix H, section H.7 details the engine-out calculations. The engines were put at a five degree cant to lessen the thrust moment arm about the C.G. This allowed

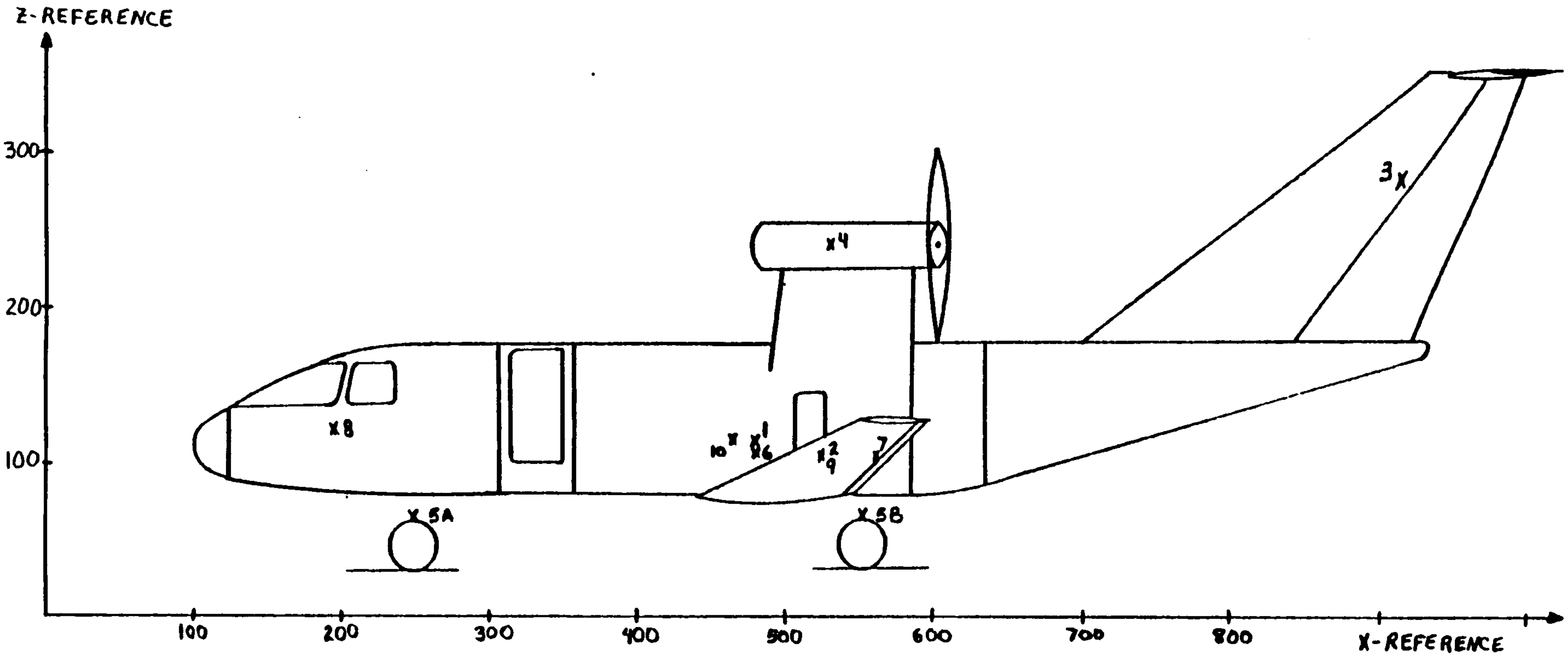

FIGURE 3.1.3 25 PASSENGER GENERAL ARRANGEMENT

TABLE 3.1.4 25 PASSENGER COMMUTER
CLASS I WEIGHT AND BALANCE CALCULATION

#	COMPONENT	W_i	x_i	$W_i x_i$	z_i	$W_i z_i$
1.	Fuselage	2526	487		120	
2.	Wing	2294	520		110	
3.	Empennage	568	920		252	
4.	Engine	2526	520		232	
5.	Nose Gear	288	250		65	
	Main Gear	575	550		65	
6.	Fixed eqpt.	2862	487		124	
Empty Weight:	W_e	= 11639		6041166	$X_{cg_{we}}$ = 519	
					$Z_{cg_{we}}$ = 146	
7.	Trp. fuel/oil	105	555		110	
8.	Crew	410	195		124	
Operating Weight Empty:	W_{OE}	= 12154		6179391	$X_{cg_{woe}}$ = 508	
					$Z_{cg_{woe}}$ = 145	
9.	Fuel	3767	520		110	
	$W_{OE} + W_F$	= 15921		8138231	$X_{cg_{woe+wf}}$ = 511	
10.	Passengers	5125	472		124	
	$W_{OE} + W_{pax}$	= 17279		8598391	$X_{cg_{woe+wpax}}$ = 498	
Take-off Weight:	W_{TO}	= 21046		10557231	$X_{cg_{wto}}$ = 502	
					$Z_{cg_{wto}}$ = 133	

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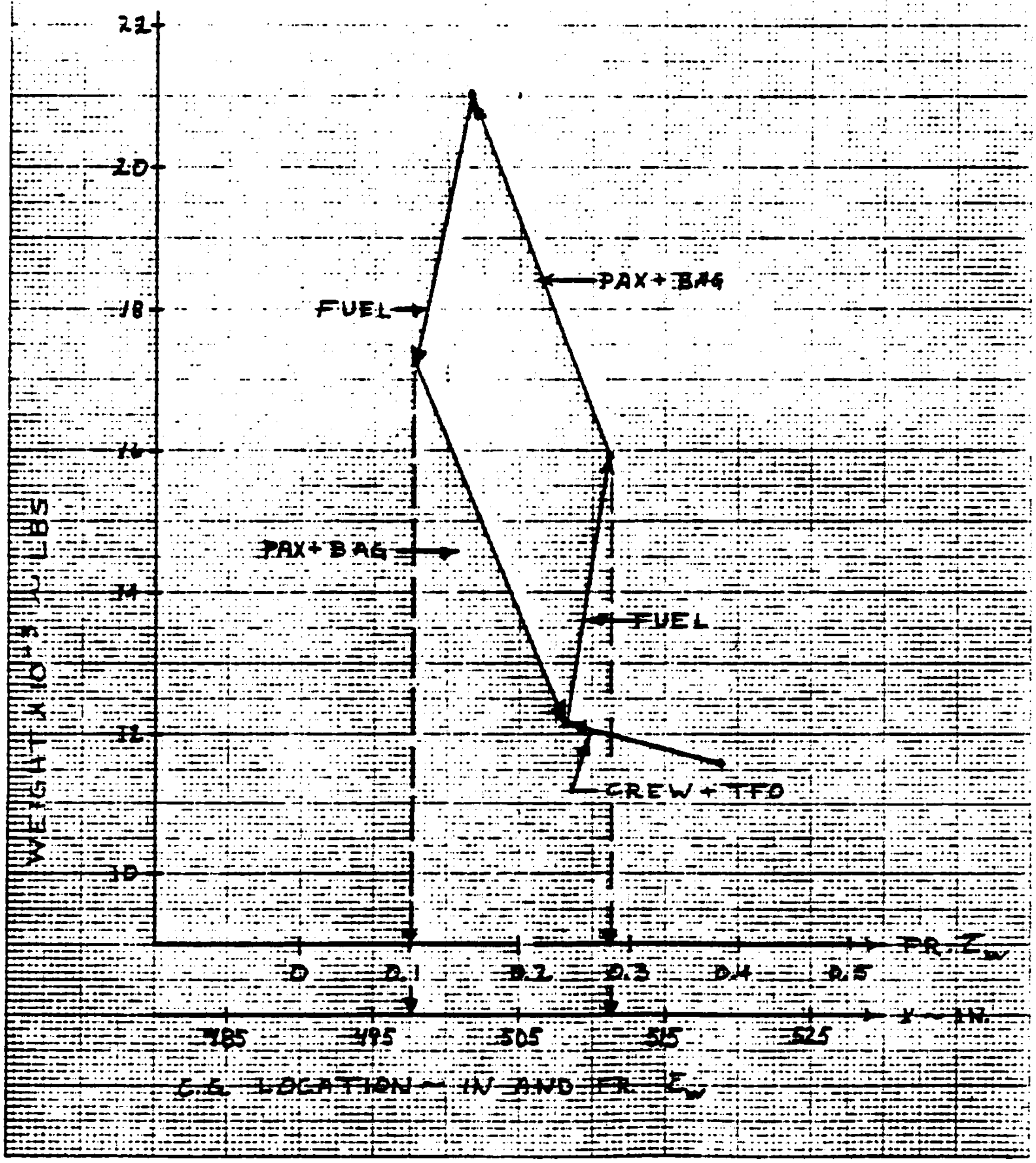

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FIGURE 3.1.4 CENTER OF GRAVITY EXCURSION DIAGRAM OF THE 25 PASSENGER MODEL

TABLE 3.1.5 STABILITY AND CONTROL RESULTS FOR THE 25 PASSENGER COMMUTER

$S = 421 \text{ ft}^2$

$\bar{c} = 6.28 \text{ ft}$ F.S. 487 = LE $\bar{c}_w$

$b = 71.1 \text{ ft}$

$S_H = 69 \text{ ft}^2$

$S_V = 170 \text{ ft}^2$

$\Delta\bar{X}_{AC_B} = -.34$

$\bar{X}_{AC_{WB}} = -.09$

$\bar{X}_{AC_A} = .45$ F.S 521

$\bar{X}_{AC_H} = 6.30$

$C_{L_{\alpha_W}} = 4.71 \text{ rad}^{-1}$

$C_{L_{\alpha_H}} = 3.41 \text{ rad}^{-1}$

$C_{L_{\alpha_V}} = 1.46 \text{ rad}^{-1}$

$C_{n_\beta} = .084 \text{ rad}^{-1}$

$\frac{d\varepsilon}{d\alpha} = .22$

$\bar{X}_{CG_{aft}} = .32$ F.S. 511

$X_V = 31.4 \text{ ft}$

*All results calculated from References 5. and 6.

for a vertical tail area of 170 ft^2. Appendix H, Figure H.3 contains a directional X-plot for the airplane. It can be seen that 170 ft^2 vertical tail yields a $c_{n_\beta} = .0015\ deg^{-1}$.

3.1.10 CLASS I DRAG POLARS

From methods in Reference 2 chapter 12. component wetted areas were calculated. See Table 3.1.6. and Appendix H, section H.8. From the total airplane wetted area and assuming a skin friction coefficient of .0025, C_{D_o} for the airplane was calculated. Table 3.1.7 contains the take-off, cruise, and landing drag polars computed during the initial performance sizing. These drag polars are compared to the drag polars computed from wetted area considerations. These class I drag polars more accurately represent the airplane. Changes to C_{D_o} for take-off and landing polars are given in Appendix H, section H.8.

TABLE 3.1.6 WETTED AREA BREAKDOWN

COMPONENT	WETTED AREA (ft^2)
Wing	717
Horizontal Tail	142
Vertical Tail	349
Fuselage	1471
Engine Nacelles	90x2
Engine Pylons	80
Total	2939

From Figure 3.21 Reference 1, assuming a $c_f = .0025$.

$$f = 7.2\ ft^2$$
$$C_{D_o} = f/S_{ref} = 7.2/421 = .0171$$

Now the drag polars can be calculated.

TABLE 3.1.7 DRAG POLAR COMPARISON

FLIGHT CONDITION	INITIAL	$(L/D)_{max}$	CLASS I	$(L/D)_{max}$
Take-off	$C_D=.0362+.0332\ C_L^2$	14.4	$C_D=.0321+.0332\ C_L^2$	15.3
Cruise	$C_D=.0162+.0312\ C_L^2$	22.2	$C_D=.0173+.0312\ C_L^2$	21.5
Landing	$C_D=.0662+.0332\ C_L^2$	10.7	$C_D=.1071+.0332\ C_L^2$	8.4

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Assuming a $C_{L_{CR}} = .3$

$$(L/D)_{CR} = 14.9$$

During initial take-off weight sizing $(L/D)_{CR}$ was assumed to be 16.

The sensitivities to W_{TO} given in Table 3.1.3 show that:

$$\frac{\partial W_{TO}}{\partial (L/D)} = -500.7 \text{ lbs}$$

Therefore for the baseline configuration:

$$\Delta(L/D)_{CR} = 14.9 - 16 = -1.1$$

$$\Delta W_{TO} = \Delta(L/D)_{CR} \frac{\partial W_{TO}}{\partial (L/D)} = 551 \text{ lbs}$$

Since W_{TO} = 21046 lbs, the reduction in $(L/D)_{CR}$ causes a 2.6% increase in W_{TO}. This small change does not warrant resizing of the airplane take-off weight.

3.2 PRELIMINARY DESIGN OF THE 36 PASSENGER BASELINE CONFIGURATION

Figure 3.2.1 contains the class I 3-view for the 36 passenger commuter. Table 3.2.1 contains the geometry of the configurations

3.2.1 INITIAL WEIGHT AND PERFORMANCE SIZING FOR THE 36 PASSENGER BASELINE CONFIGURATION

3.2.1.1 INITIAL WEIGHT SIZING

Initial weight sizing was conducted using a method in Reference 1. The following assumptions were made for the airplane:

1) $(L/D)_{cr} = 16$
2) $C_p = 0.4$ lbs/hp/hr

The above assumptions and the mission specifications, given in Table 3.2.2, yielded the airplane weights and sensitivities in Table 3.2.3. Appendix I, section I.2 contains output from XEWTOG, a computerized weight sizing method developed at the University of Kansas.

3.2.1.2 INITIAL PERFORMANCE SIZING

XPRFRM, a computer program developed at the University of Kansas, was used to determine the required take-off power, P_{TO} and wing area, S that meet the performance criteria given in Table 3.2.2. XPRFRM follows the method of Reference 1. Maximum lift coefficients and wing aspect ratio are also determined. Figure 3.2.2 shows the required power loading, wing loading combinations that satisfy the performance criteria. From Figure 3.2.2 it is determined that cruise speed and landing field length requirements are critical for this airplane. The results of the performance sizing effort are listed in Table 3.2.2. Appendix I, section I.3 details the computer output of XPRFRM.

TABLE 3.2.1 TABLE OF GEOMETRY FOR THE 36 PASSENGER COMMUTER

	WING	HORIZONTAL TAIL	VERTICAL TAIL
S ft^2	449	69	130
b ft	73.4	16.6	12
$\bar{c}$ ft	6.5	4.2	11.88
$\bar{c}$ LE F.S.	571	1080	938
A	12	4	1.1
Λ_{LE}	15°	20°	58°
λ	.4	.7	.3
t/c	.13 root	.11	.11
Airfoil	NLF	NLF (sym)	NLF (sym)
Γ	7°	0°	0°
i	3°	0°	0°
		elevator chord ratio .36	rudder chord ratio .35

Spoiler: chord ratio .12
span ratio .58 to .88

Flap: chord ratio .25
span ratio .11 to 1.0

	FUSELAGE	CABIN INTERIOR	OVERALL
Length ft	78.1	36.7	86.0
Height in	96	76	290
Width in	96	91	881

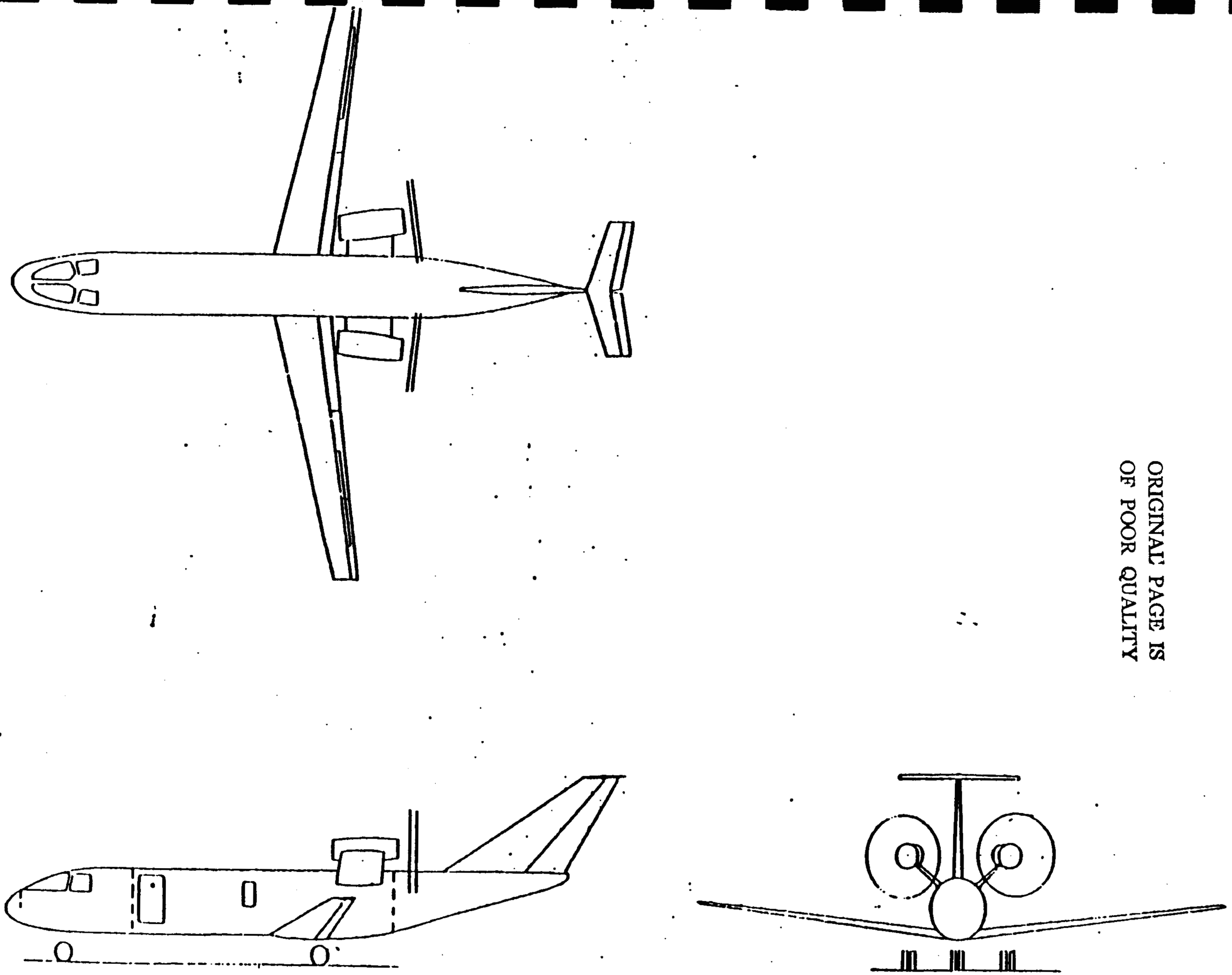


FIGURE 3.2.1 3-VIEW OF THE 36 PASSENGER MODEL

TABLE 3.2.2 MISSION SPECIFICATION FOR A 36 PASSENGER ADVANCED TECHNOLOGY COMMUTER AIRPLANE

PAYLOAD:	36 passengers at 175 lbs each with 30 lbs of baggage per passenger, carry-on luggage capability is required
CREW:	2 pilots and 1 flight attendant at 175 lbs each with 30 lbs of baggage each
RANGE:	1100 nm with maximum payload with 25% fuel reserves
ALTITUDE:	30,000 ft at the design range
CRUISE SPEED:	Mach = 0.70
CLIMB:	climb rate of 3000 fpm
TAKE-OFF AND LANDING:	3500 ft balanced field length
POWERPLANTS:	advanced turboprops
PRESSURIZATION:	5000 ft cabin at 30,000 ft
CERTIFICATION BASE:	FAR 25

MISSION PROFILE:

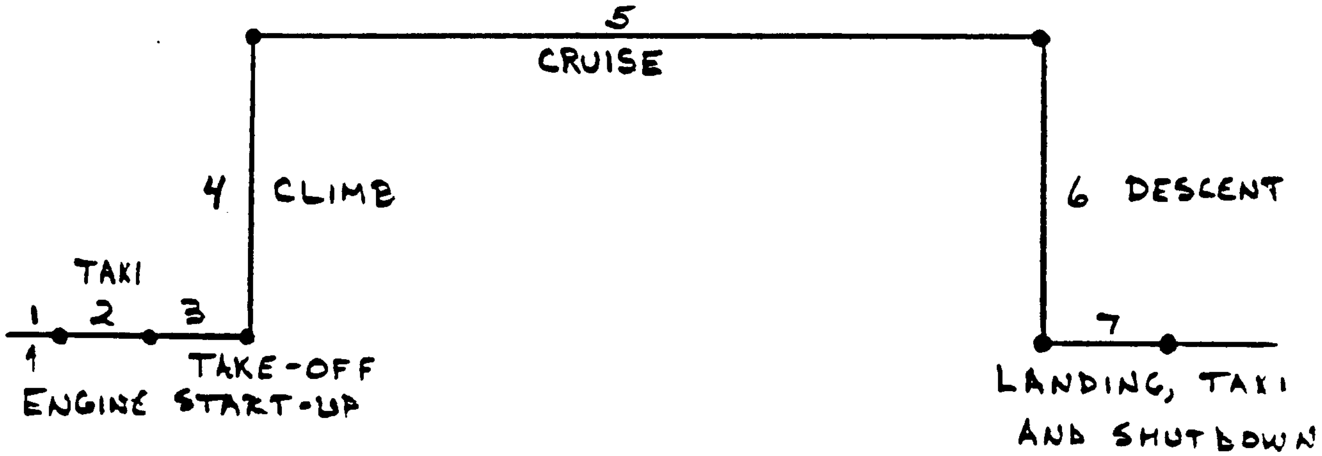


TABLE 3.2.3 INITIAL SIZING PARAMETERS FOR THE 36 PASSENGER COMMUTER

Weights: Take-off Weight - W_{TO} = 31395 lbs
Operating Weight Empty - W_{OE} = 18395 lbs
Payload Weight - W_{PL} = 7380 lbs
Crew Weight - W_{CREW} = 615 lbs
Mission Fuel Weight - W_F = 5620 lbs

Wing Area - $S = 449\ ft^2$
Wing Aspect Ratio - $A = 12$
Take-off Power - P_{TO} = 8970 shp
Required Lift Coefficients -
Clean $C_{L_{MAX}} = 1.4$
Take-off $C_{L_{MAX}} = 1.4$
Landing $C_{L_{MAX}} = 3.0$

Take-off Weight Sensitivities -

$$\frac{\partial W_{TO}}{\partial C_p} = 30976.4 \quad \text{(lb/lb/hp/hr)}$$

$$\frac{\partial W_{TO}}{\partial \eta_p} = -14577.1 \quad \text{(lbs)}$$

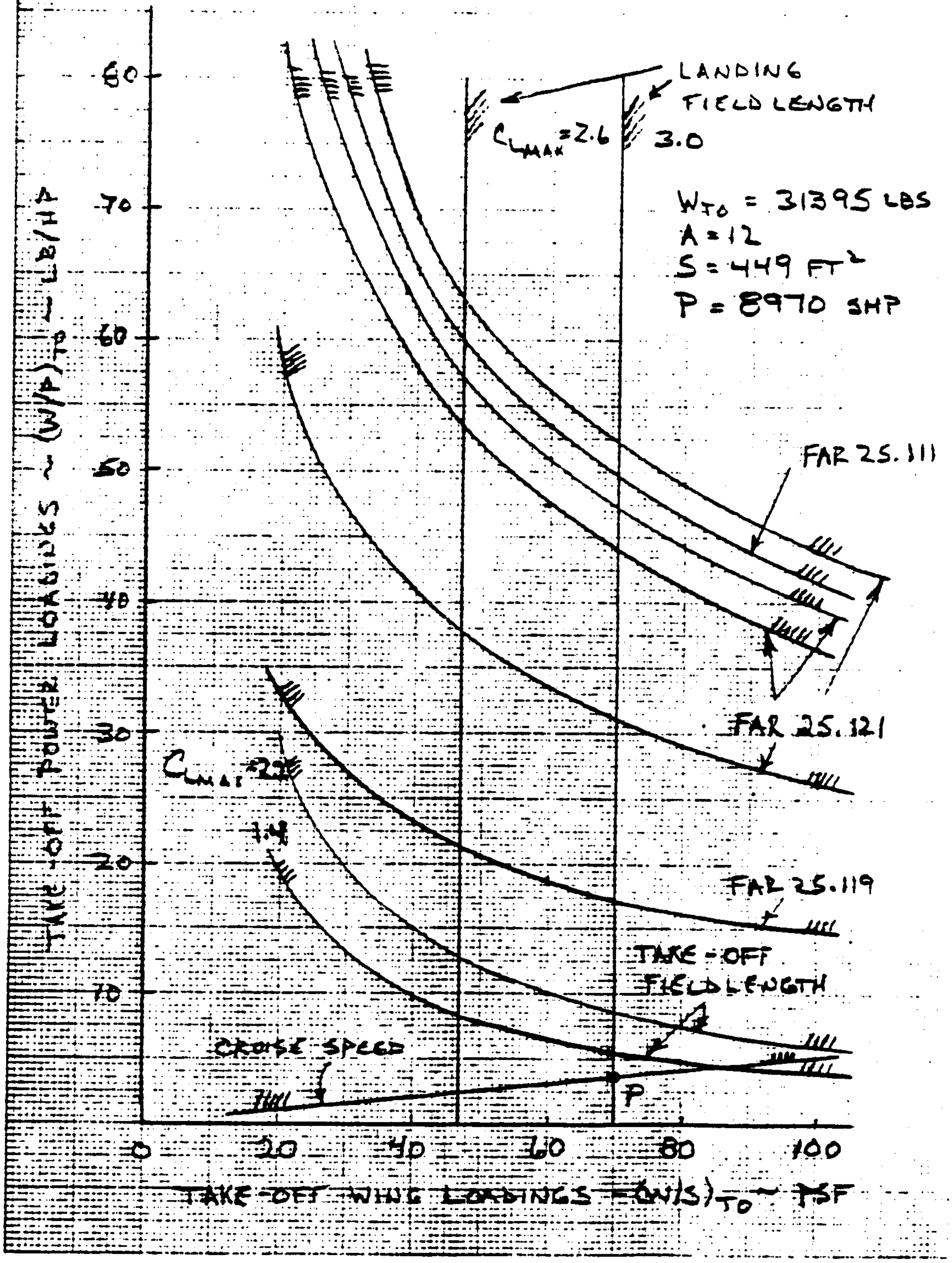
$$\frac{\partial W_{TO}}{\partial (L/D)} = -744.4 \quad \text{(lbs)}$$

$$\frac{\partial W_{TO}}{\partial R} = 11.3 \quad \text{(lb/nm)}$$

3.2.2 FUSELAGE AND COCKPIT LAYOUTS

The 36 passenger airplane has the same flight deck layout and fuselage cross section as the rest of the commuter family. The cockpit design and the fuselage cross section are contained in Appendix A. The lengths of the fuselage and cabin are given in Table 3.2.1.

The design methodology followed the steps in Reference 2. and 3.



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3.2.3 ENGINE SELECTION

The commuter family will be powered by 2 advanced turboprop engines. The 36 passenger requires the use of two 6000 shp turboprops.

Appendix B contains engine data for the airplane.

3.2.4 WING AND FLAP DESIGN

Table 3.2.1 presents the geometry of the wing and flaps. Parameters such as leading edge sweep and wing thickness were dictated by the selection of an NLF Airfoil. Appendix C contains the airfoil cross section and airfoil parameters. Wing parameters were selected using the method of Reference 2. chapter 6.

The flaps were sized to a $C_{L_{MAX_L}} = 3.0$. This required the use of fowler flaps. The sizing methods used are contained in chapter 7 of Reference 2. The design calculations are in Appendix I, section I.4.

3.2.5 DESIGN OF THE EMPENNAGE

Table 3.2.1 shows the empennage for the 36 passenger airplane. Initially the V-bar method of chapter 8 in Reference 2. was used to size the empennage. The design calculations are in Appendix I, section I.5. The initial tail areas that resulted are listed below:

$$S_H = 69 \text{ ft}^2$$

$$S_V = 78 \text{ ft}^2$$

The empennage was redesigned from stability and control considerations. These considerations are discussed in section 3.2.9.

3.2.6 CONTROL SURFACE SIZING

3.2.6.1 LATERAL - DIRECTIONAL CONTROLS

Since full span flaps were required for landing, spoilers were used in place of ailerons. The spoiler geometry was determined from chapter 8 of Reference 2. Spoiler geometry is contained in Table 3.2.1. The rudder was also sized from methods in chapter 8 of Reference 2. Its geometry is contained in Table 3.2.1.

3.2.6.2 LONGITUDINAL CONTROLS

The elevators were sized using methods in chapter 8 of Reference 2. The geometry of the elevator is contained in Table 3.2.1

3.2.7 LANDING GEAR DESIGN

From Reference 2. chapter 9. it was determined that a 30" x 9" tire could be utilized for the nose and main landing gear on every airplane of the commuter family. A preliminary retraction scheme for the main gear is shown in Appendix D. The gear placement was dictated by the weight and balance calculations shown in section 3.2.8. Lateral tip-over, and longitudinal gear placement criteria given in Reference 2. were met. Appendix I, section I.6 contains the lateral tip-over calculations.

3.2.8 CLASS I WEIGHT AND BALANCE CALCULATIONS

Class I component weights were calculated by averaging typical take-off weight fractions of commuter airplanes. Appendix F contains the class I weight fractions for the commuter family. Using methods in chapter 10 of Reference 2. A preliminary weight and balance of the 36 passenger commuter was determined. Component weights and center of gravity locations are contained in Table 3.2.4. A general arrangement drawing is contained in Figure 3.2.3. The center of gravity excursion diagram is contained in Figure 3.2.4. The 36 passenger commuter has a 22" excursion range. This is $.28\ \bar{c}_w$.

3.2.9 STABILITY AND CONTROL RESULTS

A class I stability and control analysis was performed using the methods of Reference 2. chapter 11. Table 3.2.5 contains geometric quantities and stability derivatives necessary to size the empennage from stability and control considerations. Design calculations are located in Appendix I, section I.7.

3.2.9.1 LONGITUDINAL STABILITY

From methods in chapter 11. of Reference 2. the horizontal tail was resized to incorporate a desired static margin of 5%. Appendix I, Figure I.2 presents the longitudinal X-plot for the airplane. From this plot it is seen that a tail area of 62 ft^2 is required. Since 69 ft^2 was the original estimate, it was decided that not enough area change occurred to warrant resizing the horizontal tail.

3.2.9.2 LATERAL - DIRECTIONAL STABILITY

From methods in chapter 11 of Reference 2. the vertical tail area required to hold engine-out flight was determined to be critical. Appendix I, section I.7 details the engine-out calculations. The engines were put at a five degree cant

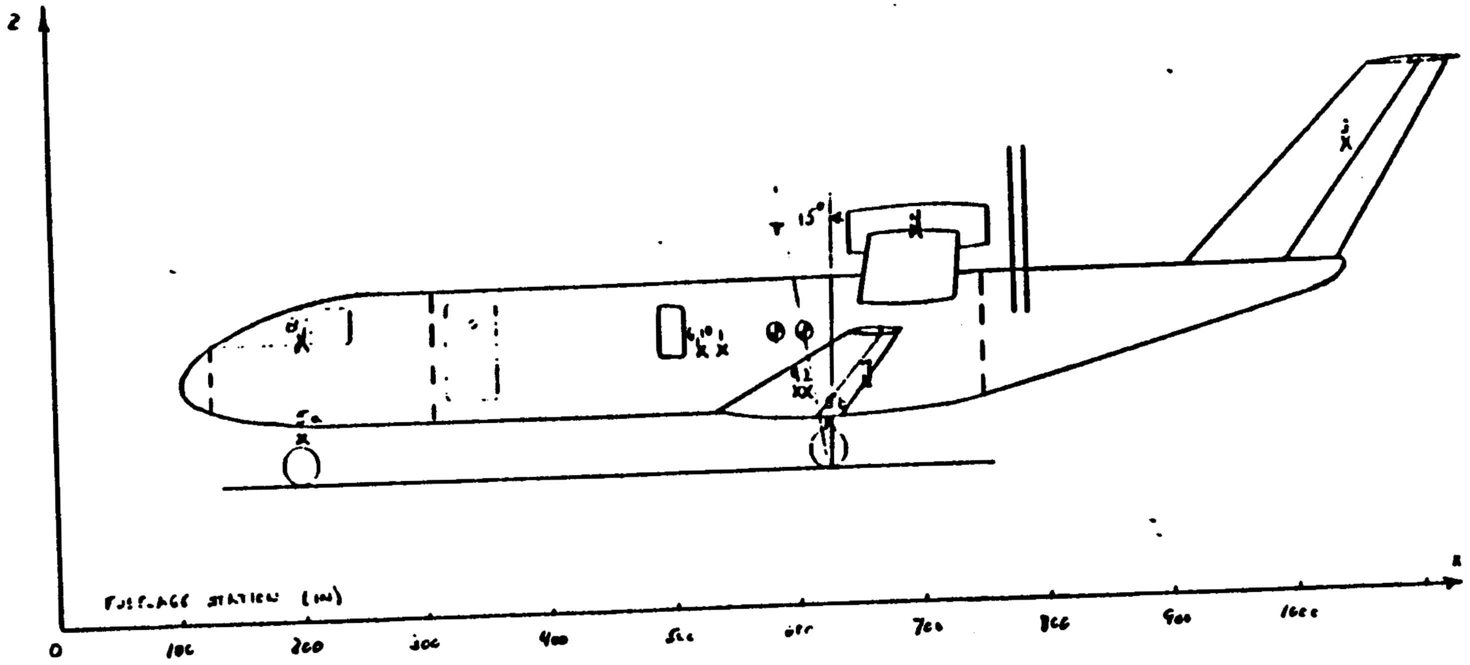

FIGURE 3.2.3 36 PASSENGER GENERAL ARRANGEMENT

TABLE 3.2.4 36 PASSENGER COMMUTER
CLASS I WEIGHT AND BALANCE CALCULATION

#	COMPONENT	W_i	x_i	$W_i x_i$	z_i	$W_i z_i$
1.	Fuselage	3767	541		191	
2.	Wing	3422	610		166	
3.	Empennage	847	1045		320	
4.	Engine	4105	700		276	
5.	Nose Gear	429	125		137	
	Main Gear	858	620		137	
6.	Fixed eqpt.	4270	525		191	
	Empty Weight: W_e =	17698		10741347	$X_{cg_{we}} = 607$ $Z_{cg_{we}} = 208$	
7.	Trp. fuel/oil	82	655		166	
8.	Crew	615	200		191	
	Operating Weight Empty: W_{OE} =	18395		10918057	$X_{cg_{woe}} = 594$ $Z_{cg_{woe}} = 207$	
9.	Fuel	5620	605		166	
	$W_{OE} + W_F$ =	24015		14318157	$X_{cg_{woe+wf}} = 596$	
10.	Passengers	7380	525		191	
	$W_{OE} + W_{pax}$ =	25775		14792557	$X_{cg_{woe+wpax}} = 574$	
	Take-off Weight: W_{TO} =	31395		18192657	$X_{cg_{wto}} = 579$ $Z_{cg_{wto}} = 196$	

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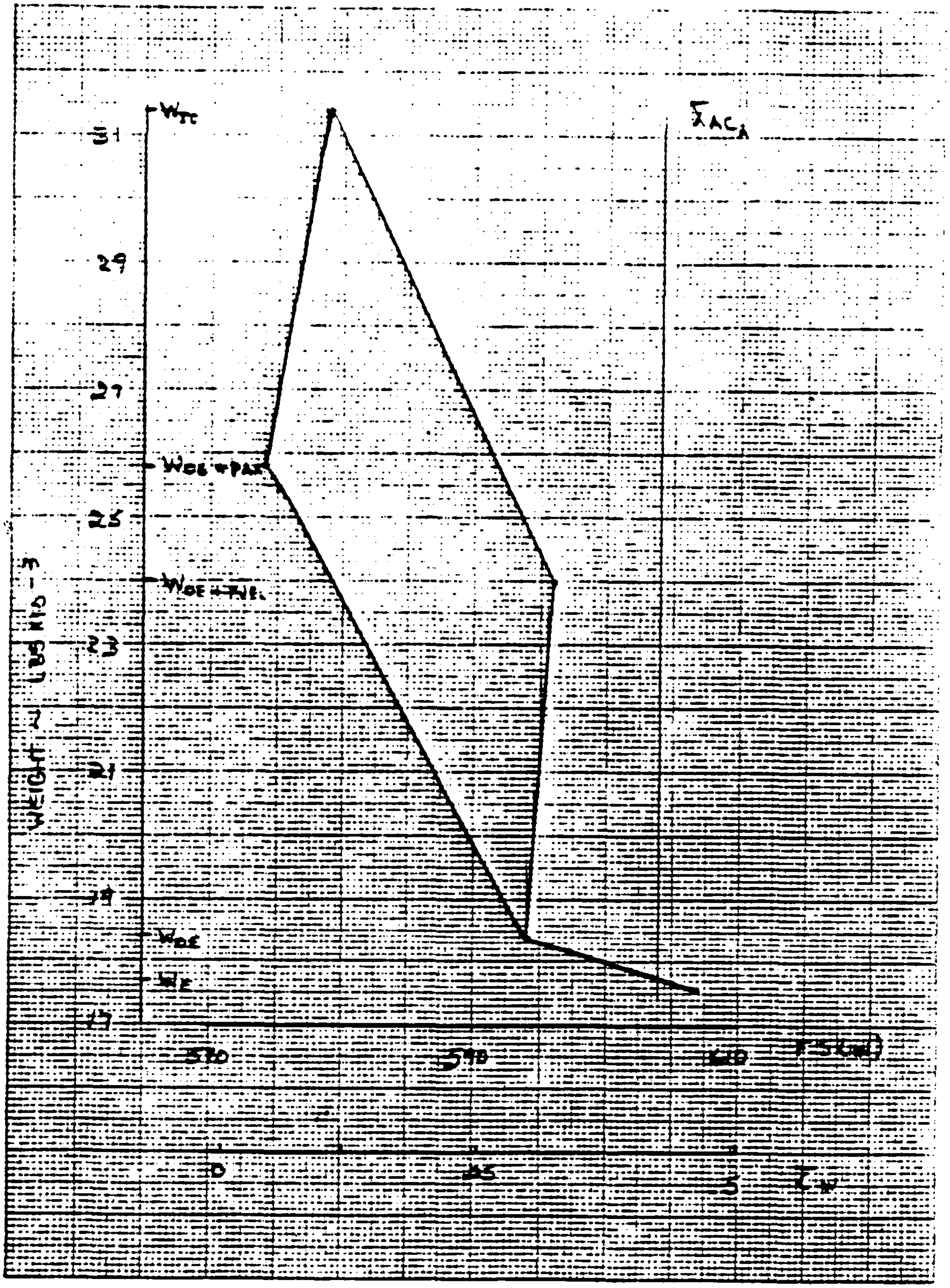

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FIGURE 3.2.4 CENTER OF GRAVITY EXCURSION DIAGRAM OF THE 36 PASSENGER MODEL

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TABLE 3.2.5 STABILITY AND CONTROL RESULTS FOR THE 36 PASSENGER COMMUTER

$S = 449 \text{ ft}^2$

$\bar{c} = 6.5 \text{ ft}$ F.S. 571 = LE $\bar{c}_w$

$b = 73.4 \text{ ft}$

$S_H = 69 \text{ ft}^2$

$S_V = 130 \text{ ft}^2$

$\Delta\bar{x}_{AC_B} = -.33$

$\bar{x}_{AC_{WB}} = -.08$

$\bar{x}_{AC_A} = .43$ F.S 604

$\bar{x}_{AC_H} = 6.40$

$C_{L_{\alpha_W}} = 4.71 \text{ rad}^{-1}$

$C_{L_{\alpha_H}} = 3.41 \text{ rad}^{-1}$

$C_{L_{\alpha_V}} = 1.46 \text{ rad}^{-1}$

$C_{n_\beta} = .178 \text{ rad}^{-1}$

$\frac{d\varepsilon}{d\alpha} = .236$

$\bar{x}_{CG_{aft}} = .33$ F.S. 597

$x_V = 34.67 \text{ ft}$

*All results calculated from References 5. and 6.

to lessen the thrust moment arm about the C.G. This allowed for a vertical tail area of 130 ft^2. Appendix I, Figure I.3 contains a directional X-plot for the airplane. It can be seen that 130 ft^2 vertical tail yields a c_{n_β} = .0030 deg^{-1}.

3.2.10 CLASS I DRAG POLARS

From methods in Reference 2 chapter 12. component wetted areas were calculated. See Table 3.2.6. and Appendix I, section I.8. From the total airplane wetted area and assuming a skin friction coefficient of .0025, C_{D_o} for the airplane was calculated. Table 3.2.7 contains the take-off, cruise, and landing drag polars computed during the initial performance sizing. These drag polars are compared to the drag polars computed from wetted area considerations. These class I drag polars more accurately represent the airplane. Changes to C_{D_o} for take-off and landing polars are given in Appendix I, section I.8.

TABLE 3.2.6 WETTED AREA BREAKDOWN

COMPONENT	WETTED AREA (ft^2)
Wing	788
Horizontal Tail	142
Vertical Tail	267
Fuselage	1702
Engine Nacelles	90x2
Engine Pylons	62x2
Total	3203

From Figure 3.21 Reference 1, assuming a c_f = .0025.

$$f = 7.8 \text{ ft}^2$$
$$C_{D_o} = f/S_{ref} = 7.8/449 = .0174$$

Now the drag polars can be calculated.

TABLE 3.2.7 DRAG POLAR COMPARISON

FLIGHT CONDITION	INITIAL	$(L/D)_{max}$	CLASS I	$(L/D)_{max}$
Take-off	C_D=.0408+.0332 C_L^2	13.6	C_D=.0324+.0332 C_L^2	15.2
Cruise	C_D=.0241+.0312 C_L^2	18.2	C_D=.0176+.0312 C_L^2	21.3
Landing	C_D=.1076+.0332 C_L^2	8.4	C_D=.1074+.0332 C_L^2	8.4

Assuming a $C_{L_{CR}} = .3$

$$(L/D)_{CR} = 14.7$$

During initial take-off weight sizing $(L/D)_{CR}$ was assumed to be 16.

The sensitivities to W_{TO} given in Table 3.2.3 show that:

$$\frac{\partial W_{TO}}{\partial (L/D)} = -744.4 \text{ lbs}$$

Therefore for the baseline configuration:

$$\Delta(L/D)_{CR} = 14.7 - 16 = -1.3$$

$$\Delta W_{TO} = \Delta(L/D)_{CR} \frac{\partial W_{TO}}{\partial (L/D)} = 968 \text{ lbs}$$

Since W_{TO} = 31395 lbs, the reduction in $(L/D)_{CR}$ causes a 3% increase in W_{TO}. This 3% change does not warrant resizing of the airplane take-off weight.

3.3 PRESENTATION OF THE 75 PASSENGER TWIN-BODY CONFIGURATION

This section presents the class I design of a 75 passenger twin-body configuration. A class I 3-view is shown in Figure 3.3.1, with the corresponding geometric data in Table 3.3.1. The most significant advantage of this configuration is commonality. Major components of the 36 passenger design are used in the 75 passenger twin-body configuration:

Common:	Fuselage
	Wing (outboard section)
	Vertical Tail
	Horizontal Tail
	Cockpit

3.3.1 INITIAL WEIGHT AND PERFORMANCE SIZING FOR THE 75 TWIN-BODY BASELINE CONFIGURATION

3.3.1.1 INITIAL WEIGHT SIZING

The weight sizing methods in Reference 1. are empirical, using data from past airplanes. Since a data base on twin-body airplanes is nearly non-existent, this method was not used. To estimate the twin-body weight the 36 passenger airplane weights were doubled. Then adjustments for specific components were made:

Wing	-1920 lbs	(lighter center section)
Engines	+260 lbs	(larger engines)
Fixed Equipment	-801 lbs	(1 cockpit)
Total Reduction	-2461 lbs	

The mission specification and a typical mission profile are given in Table 3.3.2. Mission weights and performance estimates are presented in Table 3.3.3.

3.3.2 FUSELAGE AND COCKPIT LAYOUTS

The 75 passenger twin-body configuration will use only one cockpit. The space allotted for the cockpit in the second fuselage will be replaced with passenger seats. The cockpit and fuselage cross sections are common with the other airplanes in the commuter family. These cross sections are shown in Appendix A. Fuselage and cabin dimensions are given in Table 3.3.1.

3.3.3 ENGINE SELECTION

The twin-body configuration had the possibility of using 3 engines. However, a suitable engine arrangement with 3 engines was not found, so 2 larger engines were used. Using

TABLE 3.3.1 TABLE OF GEOMETRY FOR THE 75 PASSENGER TWIN-BODY CONFIGURATION

	WING	HORIZONTAL TAIL	VERTICAL TAIL
S ft^2	722	2x102	2x130
b ft	104.5	22.6	12
$\bar{c}$ ft	7.5	4.68	11.88
$\bar{c}$ LE F.S.	571	1080	938
A	15.1	5.0	1.1
Λ_{LE}	15°	25°	58°
λ	.4	.50	.3
t/c	.13 root	.11	.11
Airfoil	NLF	NLF (sym)	NLF (sym)
Γ	7°	0°	0°
i	-	0°	0°
		elevator chord ratio .36	rudder chord ratio .35

Spoiler: chord ratio .12
span ratio .58 to .88 (outboard section)

Flap: chord ratio .25
span ratio .11 to 1.0 (outboard section)

	FUSELAGE	CABIN INTERIOR	OVERALL
Length ft	78.1	36.7	86.0
Height in	96	76	290
Width in	96	91	881

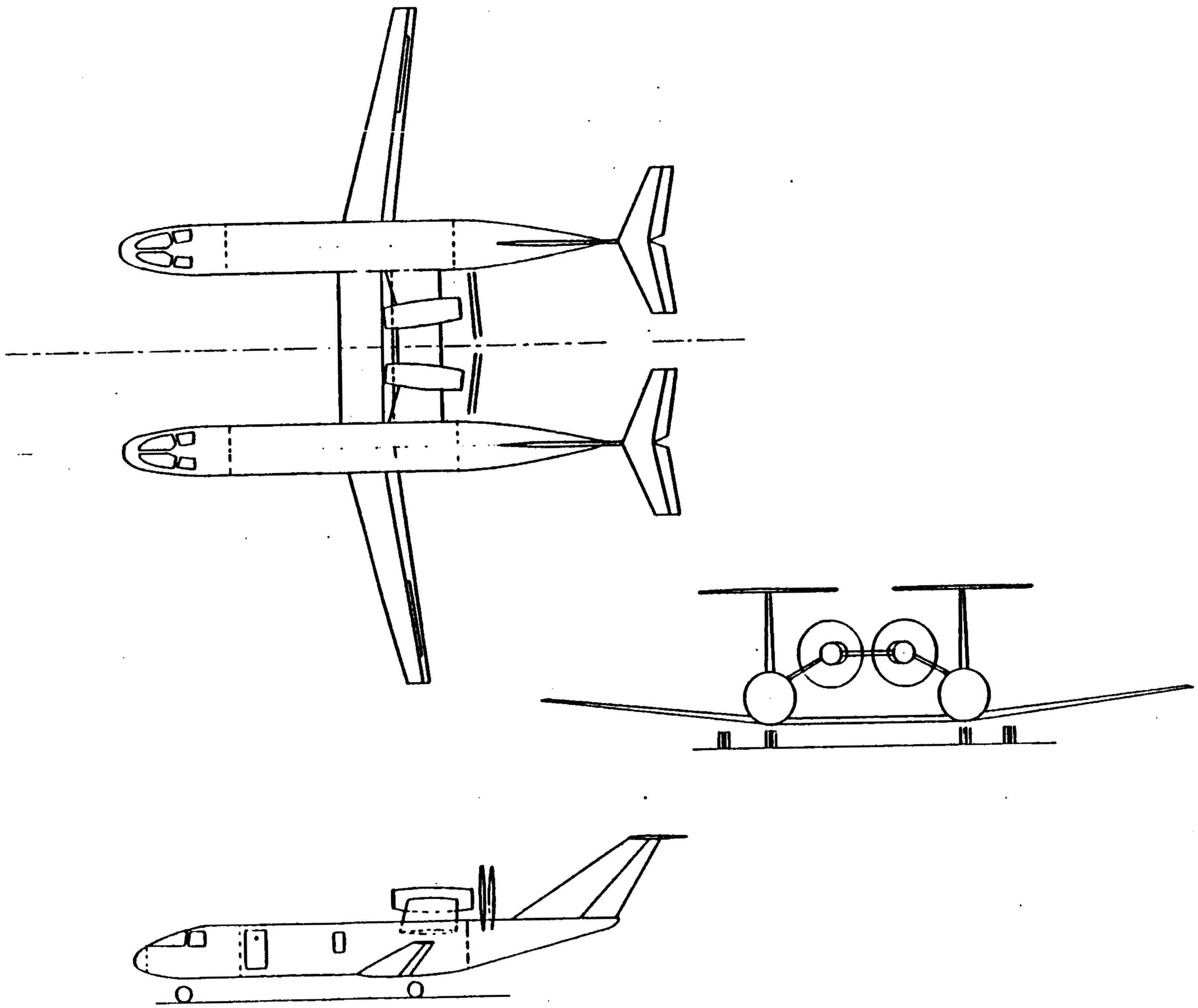

FIGURE 3.3.1 3-VIEW OF THE 75 TWIN-BODY MODEL

TABLE 3.3.2 MISSION SPECIFICATION FOR A 75 PASSENGER ADVANCED TECHNOLOGY COMMUTER AIRPLANE

PAYLOAD:	75 passengers at 175 lbs each with 30 lbs of baggage per passenger, carry-on luggage capability is required
CREW:	2 pilots and 2 flight attendants at 175 lbs with 30 lbs of baggage each
RANGE:	1500 nm with maximum payload and 25% fuel reserves
ALTITUDE:	30,000 ft at the design range
CRUISE SPEED:	Mach .70
CLIMB:	climb rate of 3000 fpm
TAKE-OFF AND LANDING:	3500 ft balanced field length
POWERPLANTS:	Advanced turboprops
PRESSURIZATION:	5000 ft cabin at 30,000 ft
CERTIFICATION BASE:	FAR 25

MISSION SPECIFICATION:

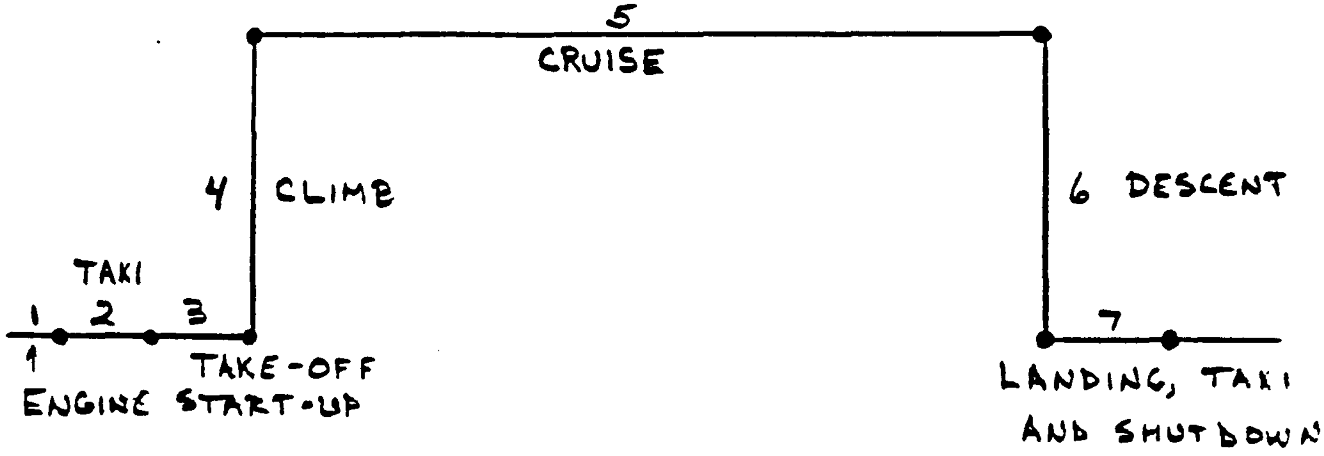

TABLE 3.3.3 INITIAL SIZING PARAMETERS FOR THE 75 PASSENGER TWIN-BODY CONFIGURATION

Weights: Take-off Weight - W_{TO} = 60683 lbs
Operating Weight Empty - W_{OE} = 34068 lbs
Payload Weight - W_{PL} = 15375 lbs
Crew Weight - W_{CREW} = 820 lbs
Mission Fuel Weight - W_F = 11240 lbs

Wing Area - $S = 722\ ft^2$
Wing Aspect Ratio - $A = 15.1$
Take-off Power - P_{TO} = 18000 shp
Required Lift Coefficients -
Clean $C_{L_{MAX}} = 1.4$
Take-off $C_{L_{MAX}} = 1.4$
Landing $C_{L_{MAX}} = 3.0$

two engines also improves the possibility of complete cockpit commonality and pilot cross rating. Two 13500 shp engines will be used. Data for these engines is contained in Appendix B.

3.3.4 WING AND FLAP DESIGN

The wing of the 75 passenger twin-body may be broken into 2 outboard sections, and an inboard section. The two outboard sections are identical to the wing for the 36 passenger airplane (see section 3.2.4). The inboard section is a straight wing that joins the two fuselages at the wing boxes. This section also transmits loads, and damps vibrations, between the two fuselages.

To achieve a high lift coefficient for landing, full span fowler flaps along both inboard and outboard wings will be required.

Data for the outboard wings (36 passenger) are given in Table 3.2.1. The 75 passenger twin-body wing data is presented in Table 3.3.1. Appendix C contains airfoil section data for the NLF airfoil.

3.3.5 DESIGN OF THE EMPENNAGE

The empennage designed for the 36 passenger airplane will be used on each fuselage of the 75 passenger twin-body. This will increase the commonality between the two airplanes. Stability and control considerations for the 75 passenger

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twin-body may require further modifications to the empennage, which are discussed in section 3.3.9.

3.3.6 CONTROL SURFACE SIZING

3.3.6.1 LATERAL - DIRECTIONAL CONTROLS

The lateral-directional controls used on the 36 passenger wing (spoilers) will also be used on the outboard 75 passenger twin-body wings. Although the moment of inertia for the twin-body is much greater, the distance of the spoilers from the C.G. is also larger. Additional lateral-directional control power may be required. Increasing the spoiler span may solve this problem. Spoiler geometries are given in Table 3.3.1.

3.3.6.2 LONGITUDINAL CONTROLS

The elevators used on the 36 passenger airplane will also be used on each of the horizontal tails. Elevator geometry is presented in Table 3.3.1.

3.3.7 LANDING GEAR DESIGN

As with the rest of the commuter family, a 30"x9" tire will be used for both main and nose gears. The gear location will be common with the 36 passenger airplane to retain commonality. Since the main gears are far from the C.G., lateral tip-over is not a concern. A gear retraction scheme is shown in Appendix D.

3.3.8 CLASS I WEIGHT AND BALANCE CALCULATIONS

A class I weight and balance calculation was done using the method of chapter 10 in Reference 2. The component weight estimates are listed in Table 3.3.4. Figure 3.3.2 shows the general arrangement and C.G. locations of the components in Table 3.3.4. There is a 23.7" (.26 $\bar{c}_w$) C.G. travel range between W_{OE} and $W_{OE} + W_{pax}$. The C.G. excursion diagram is shown in Figure 3.3.3.

3.3.9 STABILITY AND CONTROL RESULTS

Table 3.3.5 contains the geometric quantities and stability derivatives used in the stability and control calculations. The methods of chapter 11 in Reference 2 were used for the class I calculations. The design calculations are located in section M.2 of Appendix M.

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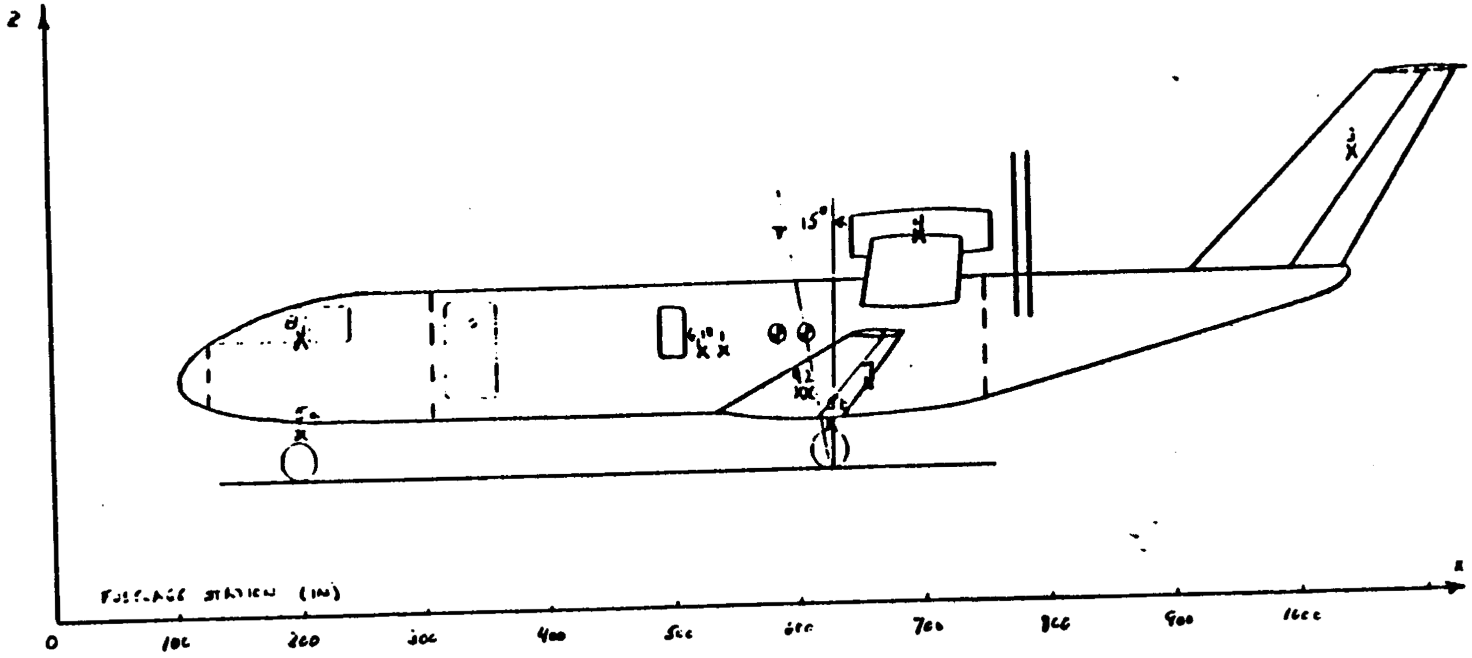

FIGURE 3.3.2 75 TWIN-BODY GENERAL ARRANGEMENT

TABLE 3.3.4 75 PASSENGER TWIN BODY
CLASS I WEIGHT AND BALANCE CALCULATION

#	Component	W_i	x_i	$W_i x_i$	z_i	$W_i z_i$
1.	Fuselage	7534	541	4076000	191	1440000
2.	Wing	4923	610	3003000	166	820000
3.	Empennage	1695	1045	1771000	320	540000
4.	Engine	8470	700	5929000	276	2340000
5.	Nose Gear	858	195	167000	137	117000
	Main Gear	1716	640	1098000	137	350000
6.	Fixed eqpt.	7739	525	4063000	191	1480000
	Empty Weight: W_e =	32935		20107000	$X_{cg_{we}}$ = 610.5 $Z_{cg_{we}}$ = 211	
7.	Trp. fuel/oil	313	655	210000	166	50000
8.	Crew	820	200	160000	191	160000
	Operating Weight Empty: W_{OE} =	34068		20477000	$X_{cg_{woe}}$ = 601 $Z_{cg_{woe}}$ = 210	
9.	Fuel	11240	630	7080000	166	1870000
	$W_{OE} + W_F$ =	45308		27557000	$X_{cg_{woe+wf}}$ = 608	
10.	Passengers	15375	525	8070000	191	2940000
	$W_{OE} + W_{pax}$ =	49443		28547000	$X_{cg_{woe+wpax}}$ = 577	
	Take-off Weight: W_{TO} =	60683		35627000	$X_{cg_{wto}}$ = 587 $Z_{cg_{wto}}$ = 198	

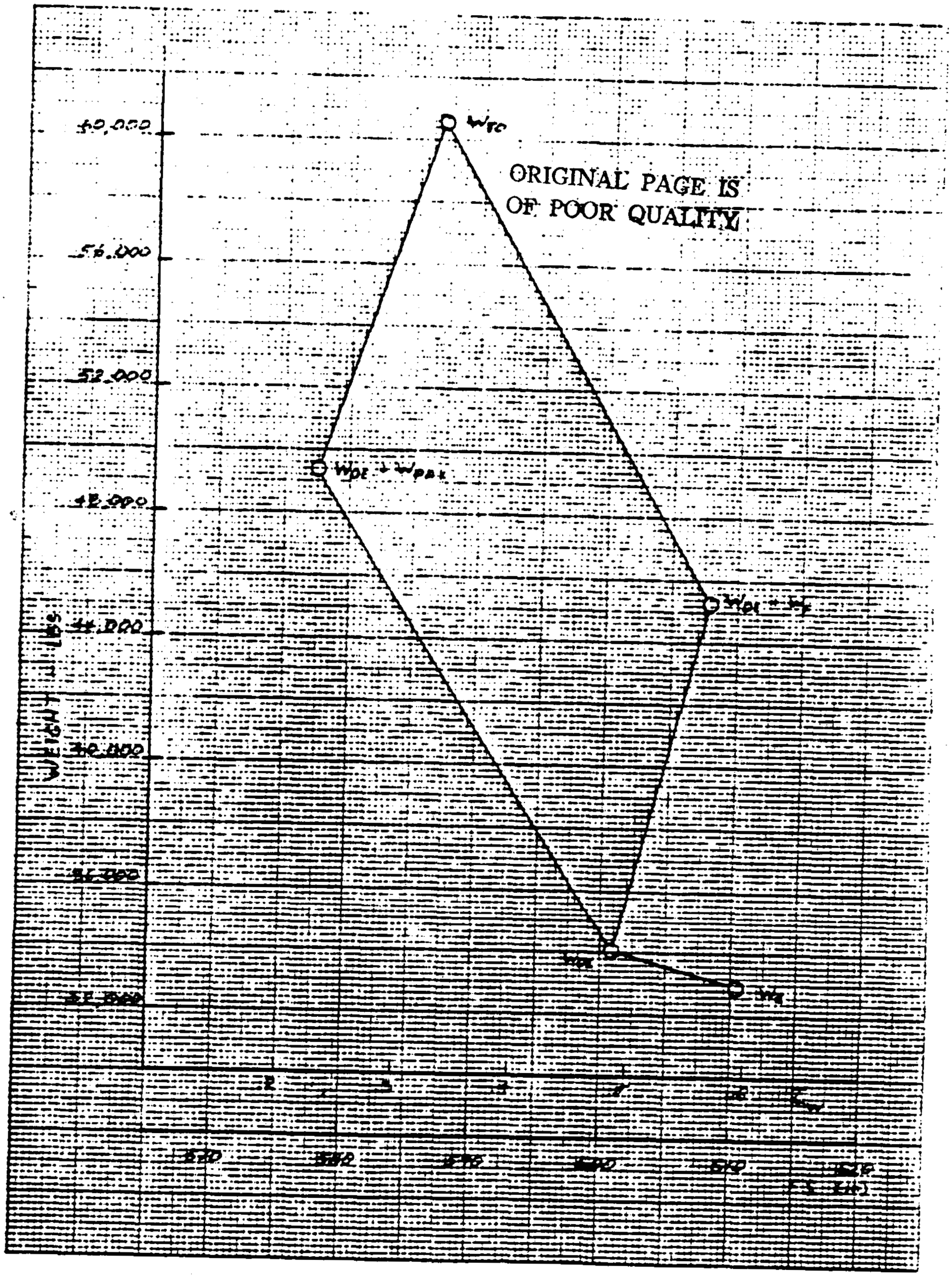

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FIGURE 3.3.3 CENTER OF GRAVITY EXCURSION DIAGRAM OF THE 75 TWIN-BODY MODEL

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3.3.9.1 LONGITUDINAL STABILITY

It was originally envisioned that the horizontal tail of the 36 passenger airplane could be used on the 75 twin-body configuration. However, from stability and control calculations using the methods of chapter 11 in Reference 2, this was not possible. These calculations and the corresponding X-plot are located in section M.2 of Appendix M. From the X-plot, a 5% static margin would require a horizontal tail area of 190 ft^2. To preserve commonality, two 102 ft^2 horizontal tails from the 50 passenger airplane will be used.

3.3.9.2 LATERAL - DIRECTIONAL STABILITY

Using the method of chapter 11 in Reference 2, the engine out for the 75 passenger twin-body is critical for the vertical tail sizing (see Appendix M, section M.2). If the vertical tails designed for the 36 passenger airplane are used, a 27° rudder deflection is required to hold engine out. From the directional X-plot located in Appendix M, Figure M.3 a total vertical tail area of 260 ft^2 (2x130) produces a $C_{n_\beta} = .0018\ deg^{-1}$.

3.3.10 CLASS I DRAG POLARS

The component wetted areas were calculated using the method of chapter 12 in Reference 2, and are listed in Table 3.3.6. A skin friction coefficient of f = .0025 is assumed. The increments in C_{D_o} due to flaps, gear, and compressibility are identical to those used in section 3.2.10. Table 3.3.7 lists the drag polars for take-off, cruise, and landing computed for this configuration. The engineering calculation for the drag polars are located in Appendix M, section M.3.

Assuming 40% of the take-off fuel weight has been used, the cruise lift coefficient is $C_{L_{cr}} = 0.36$. The lift to drag ratio is then:

$$(L/D)_{cr} = 15.3$$

From Figure 3.21 Reference 1, assuming a $c_f = .0025$.

$$f = 14.5\ ft^2$$

$$C_{D_o} = f/S_{ref} = 14.5/722 = .0201$$

TABLE 3.3.5 STABILITY AND CONTROL RESULTS FOR THE 75 PASSENGER TWIN-BODY CONFIGURATION

$S = 722\ ft^2$

$\bar{c} = 7.5\ ft$

$b = 104.5\ ft$

L.E. $\bar{c}$ = F.S. 556

$S_H = 200\ ft^2$

$S_V = 260\ ft^2$

$\Delta \bar{X}_{AC_B} = -.39$

$\bar{X}_{AC_{WB}} = -.14$

$\bar{X}_{AC_A} = .404$ F.S 592

$\bar{X}_{AC_H} = 5.77$

$C_{L_{\alpha_W}} = 4.99\ rad^{-1}$

$C_{L_{\alpha_H}} = 3.65\ rad^{-1}$

$C_{L_{\alpha_V}} = 1.46\ rad^{-1}$

$C_{n_\beta} = .102\ rad^{-1}$

$\frac{d\varepsilon}{d\alpha} = .32$

$\bar{X}_{CG_{aft}} = .58$ F.S. 608

$X_V = 34.67\ ft$

*All results calculated from References 5. and 6.

TABLE 3.3.6 WETTED AREA BREAKDOWN

COMPONENT	WETTED AREA (ft^2)
Wing	1006
Horizontal Tail	420
Vertical Tail	534
Fuselage	3404
Engine Nacelles	248
Engine Pylons	480
Total	6092

TABLE 3.3.7 DRAG POLAR COMPARISON

CONDITION	CLASS I	$(L/D)_{max}$
Take-off	$C_D = .0351 + .0264\ C_L^2$	16.4
Cruise	$C_D = .0203 + .0248\ C_L^2$	22.3
Landing	$C_D = .1101 + .0264\ C_L^2$	9.3

3.4 PRESENTATION OF THE 50 PASSENGER CONFIGURATION

Figure 3.4.1 contains the Class I 3-view for the 50 passenger commuter. Table 3.4.1 contains the geometry of the configuration.

3.4.1 Initial Sizing of the 50 Passenger Commuter

From the methods in Reference 1, the weights and initial performance parameters were selected. These parameters depended on the mission specifications. These specifications and mission profile are shown in Table 3.4.2. The following assumptions were made for the airplane:

1) $(L/D)_{cr} = 16$

2) $C_p = 0.4$ lbs/hp/hr

The preliminary weight and performance sizing are done through the use of two computer programs developed at the University of Kansas. Appendix J, Section J.2 contains output from XEWTOG, the weight sizing program. Section J.3 contains output from XPRFRM, the performance program. The results of the initial weight and performance sizing are given in Table 3.4.3. A performance matching graph is displayed in Figure 3.4.2.

3.4.2 Fuselage and Cockpit Layout

The 50 passenger airplane has the same cockpit and fuselage cross section as the rest of the commuter family. The cockpit design and fuselage cross section are contained in Appendix A. The lengths of the fuselage and cabin are given in Table 3.4.1. The design methodology followed the steps in References 2 and 3.

3.4.3 Engine Selection

The commuter family will be powered by 2 advanced turboprop engines. The 50 passenger airplane requires the use of 6000 shp turboprops. Appendix B contains the engine data used.

3.4.4 Wing and Flap Design

Table 3.4.1 presents the geometry of the wing and flaps. Parameters such as leading edge sweep and thickness were dictated by the selection of a natural laminar flow (NLF) airfoil. Appendix C contains the airfoil cross section and airfoil parameters. Wing parameters were selected using the methods of Reference 2, Chapter 6.

The flaps were sized to a $C_{L_{max_L}} = 3.0$. This required the use of Fowler flaps. The sizing methods used are contained in Chapter 7 of Reference 2. The design calculations are given in Appendix J, Section J.4.

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Table 3.4.1 Geometric Characteristics of the 50 Passenger Commuter Airplane

	Wing	Horizontal Tail	Vertical Tail
Area	592 ft^2	102 ft^2	170 ft^2
Span	84.3 ft	22.6 ft	15.4 ft
MGC	7.46 ft	4.68 ft	11.4 ft
MGC L.E.: F.S.	53.5 ft	99.9 ft	90.3 ft
Aspect Ratio	12	5.0	1.4
Sweep Angle	13.1 deg (c/4)	25.0 deg (L.E.)	40.0 deg (L.E.)
Taper Ratio	0.40	0.50	0.50
Thickness Ratio	0.13 root 0.10 tip	0.12 root 0.10 tip	0.13 root 0.12 tip
Airfoil:	All airfoils are Natural Laminar Flow airfoils.		
Dihedral Angle	7.0 deg	0 deg	0 deg
Incidence Angle	0 deg	0 deg	0 deg
Aileron chord ratio	0.25	Elevator chord ratio = 0.36 (full span)	Rudder chord ratio = 0.34 (full span)
Aileron span ratio	0.738 - 1.00		
Flap chord ratio	0.25		
Flap span ratio	0.10 - 0.738		
	Fuselage	**Cabin Interior**	**Overall**
Length	94.6 ft	45.0 ft	104 ft
Maximum Height	8.05 ft	6.30 ft	28.0 ft
Maximum Width	8.05 ft	7.60 ft	84.3 ft

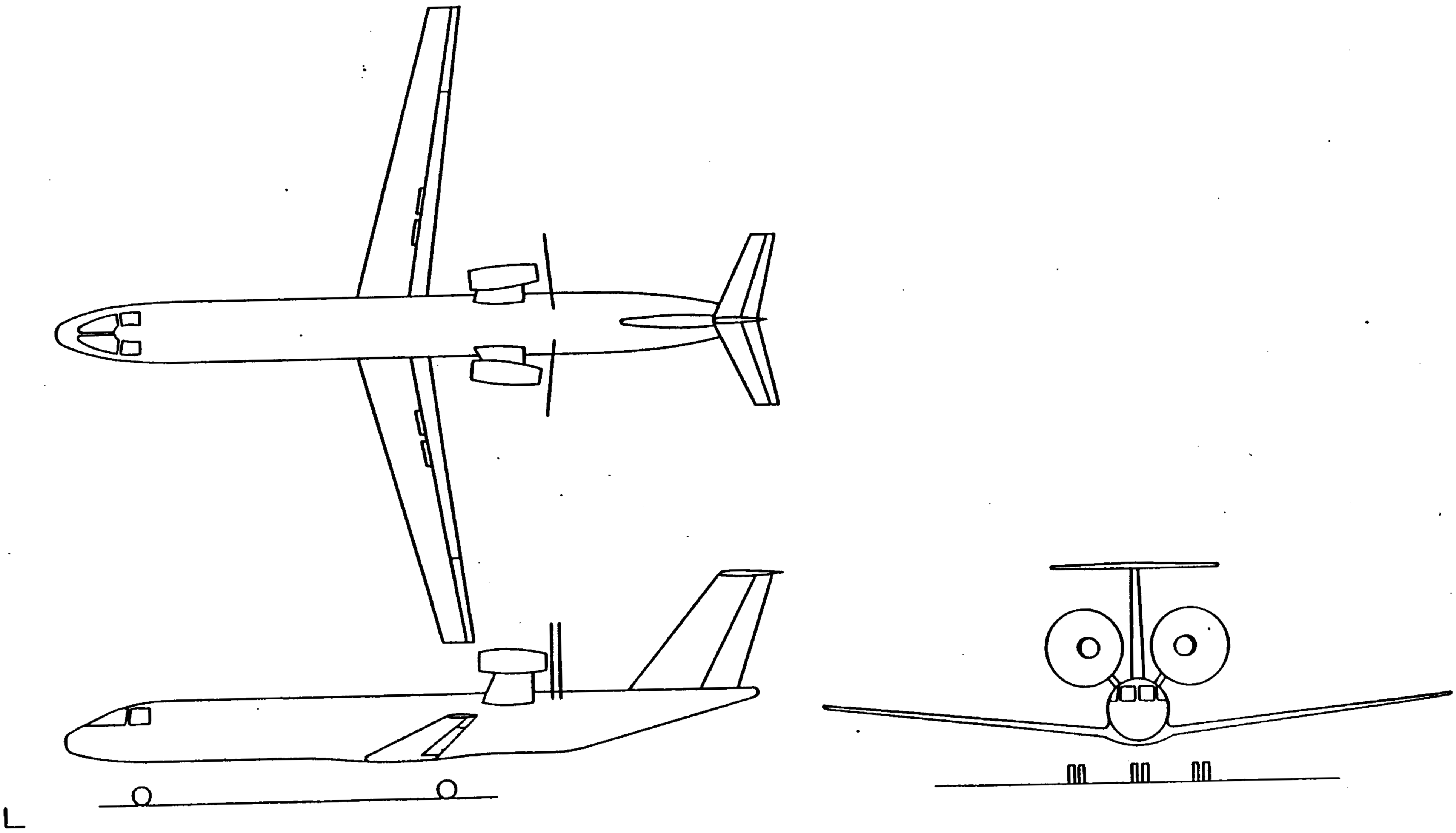


FIGURE 3.4.1 CLASS I THREE-VIEW FOR THE 50 PASSENGER AIRPLANE

Table 3.4.2 Mission Specification for the 50 Passenger Advanced Technology Commuter Airplane

PAYLOAD:	50 passengers at 175 lbs each with 30 lbs of baggage per passenger, carry-on luggage capability is required
CREW:	2 pilots and 1 flight attendant at 175 lbs each with 30 lbs of baggage each
RANGE:	1100 nm with max payload with 25% fuel reserves
ALTITUDE:	30,000 ft at the design range
CRUISE SPEED:	MACH = .70
CLIMB:	climb rate of 3000 fpm
TAKE-OFF AND LANDING:	3500 ft balanced field length
POWERPLANTS:	advanced turboprops
PRESSURIZATION:	5000 ft cabin at 30000 ft
CERTIFICATION BASE:	FAR 25

MISSION PROFILE:

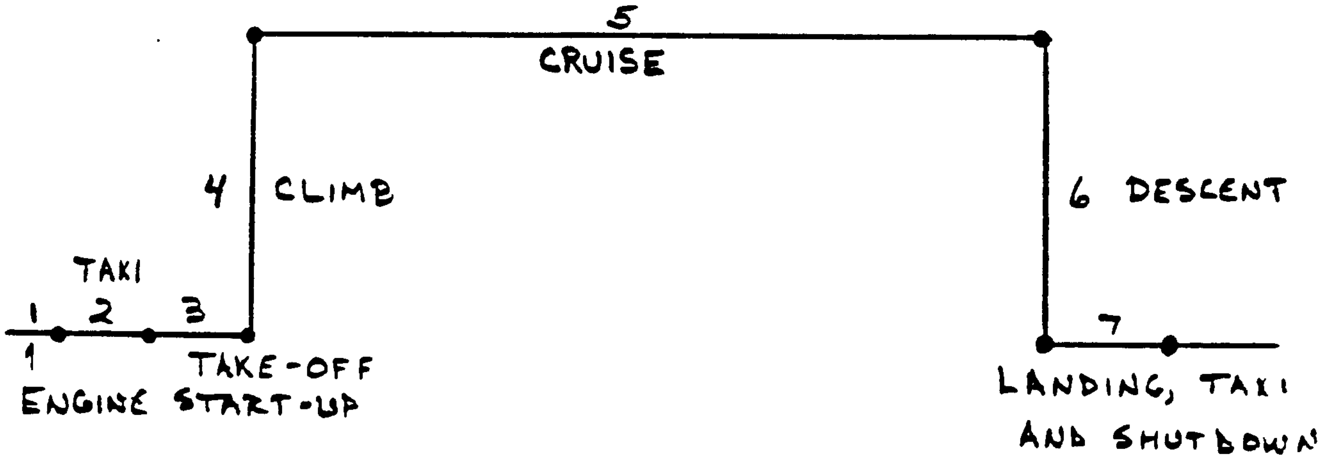


Table 3.4.3 Initial Sizing Parameters for the 50 Passenger Commuter

Weights:	Take-off Weight	W_{TO}	= 42,057 lbs
	Operating Weight Empty	W_{OE}	= 23,963 lbs
	Payload Weight	W_{PL}	= 10,250 lbs
	Crew Weight	W_{CREW}	= 615 lbs
	Mission Fuel Weight	W_F	= 6,913 lbs

Wing Area $S = 592\ ft^2$

Aspect Ratio $A = 12.0$

Take-off Power $P_{TO} = 11{,}000$ shp

Required Lift Coefficients:

Clean	$C_{L_{max}}$	= 1.5
Take-off	$C_{L_{max_{TO}}}$	= 2.0
Landing	$C_{L_{max_L}}$	= 3.0

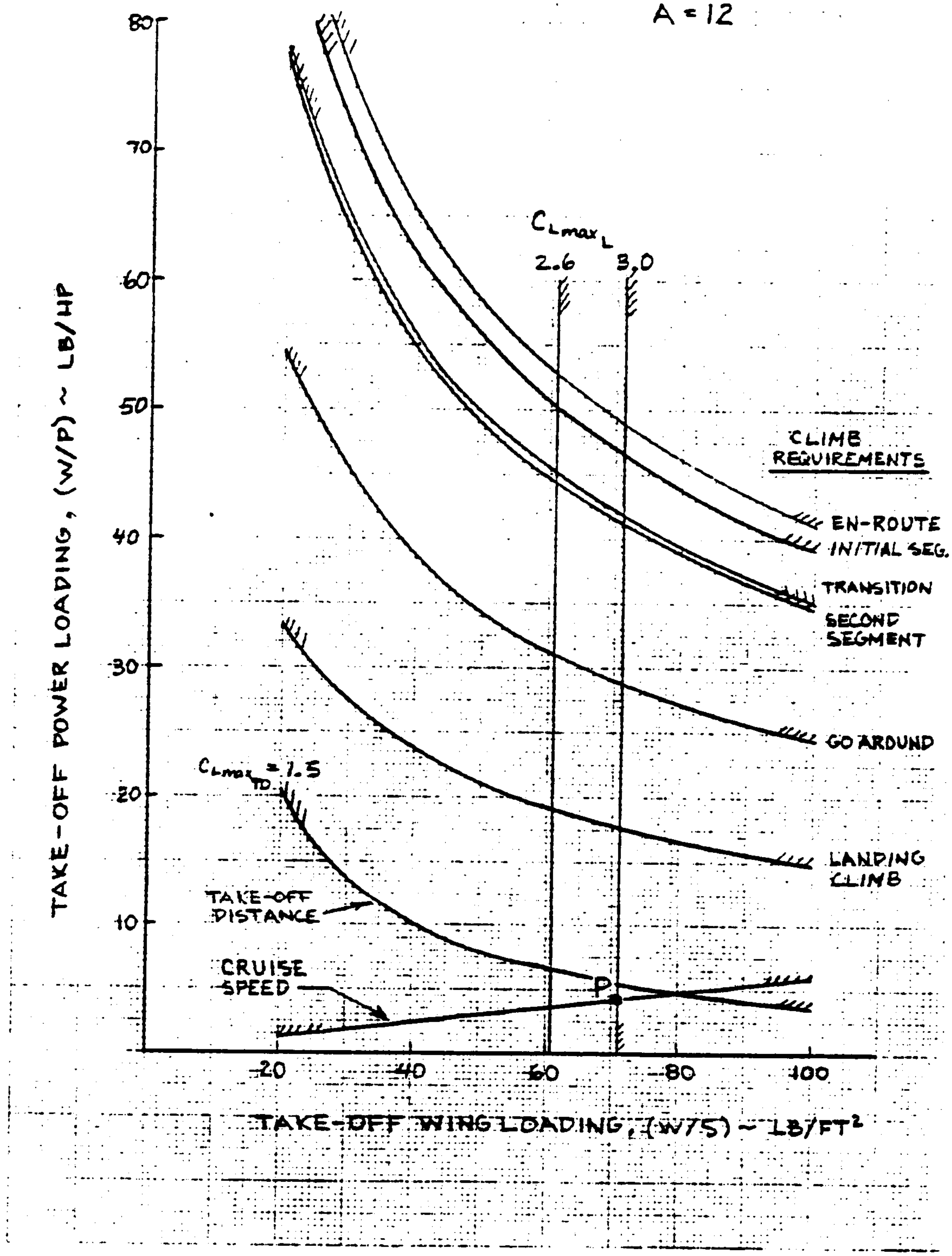
Take-off Weight Sensitivities:

$\partial W_{TO} / \partial c_p = 39{,}784$ lb/lb/hp/hr

$\partial W_{TO} / \partial \eta_p = -18{,}722$ lbs

$\partial W_{TO} / \partial (L/D) = -994.6$ lbs

$\partial W_{TO} / \partial R = 15.1$ lbs



CALC	M. RUSSELL	10-14	REVISED	DATE	FIGURE 3.4.2 PERFORMANCE MATCHING OF THE 50 PASSENGER MODEL	AE 790
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3.4.5 Design of the Empennage

Table 3.4.1 lists the empennage geometry for the 50 passenger airplane. Initially, the V-bar methods of Reference 2, Chapter 8, were used to size the empennage. These initial areas are listed below:

$$S_H = 130 \text{ ft}^2$$

$$S_V = 130 \text{ ft}^2$$

The empennage was redesigned from stability and control considerations which are discussed in section 3.4.9.

3.4.6 Control Surface Sizing

3.4.6.1 Lateral-Directional Controls

Table 3.4.1 presents the aileron geometry used. The methods used were that of Reference 2, Chapter 8.

3.4.6.2 Longitudinal Controls

The elevators were sized using methods in Chapter 8, Reference 2, and the geometry is summarized in Table 3.4.1.

3.4.7 Landing Gear Design

From Chapter 9, Reference 2, it was determined that a 30 X 9 inch tire could be used on every airplane of the commuter family. A preliminary retraction scheme for the main gear is shown in Appendix D. The gear placement was dictated by the weight and balance calculations shown in Section 3.4.8. Lateral tip-over and longitudinal gear retraction criteria given in Reference 1 were met. Appendix J, Section J.6 contains the lateral tip-over calculations.

3.4.8 Class I Weight and Balance Calculations

A preliminary weight and balance of the 50 passenger commuter was determined by using methods in Reference 2, Chapter 10. Component weights and center of gravity locations are contained in Table 3.4.4. A general arrangement drawing is provided by Figure 3.4.3. The weight-center of gravity excursion diagram is contained in Figure 3.4.4. The 50 passenger commuter has a 15 inch excursion range which corresponds to 0.17 $\bar{c}_w$.

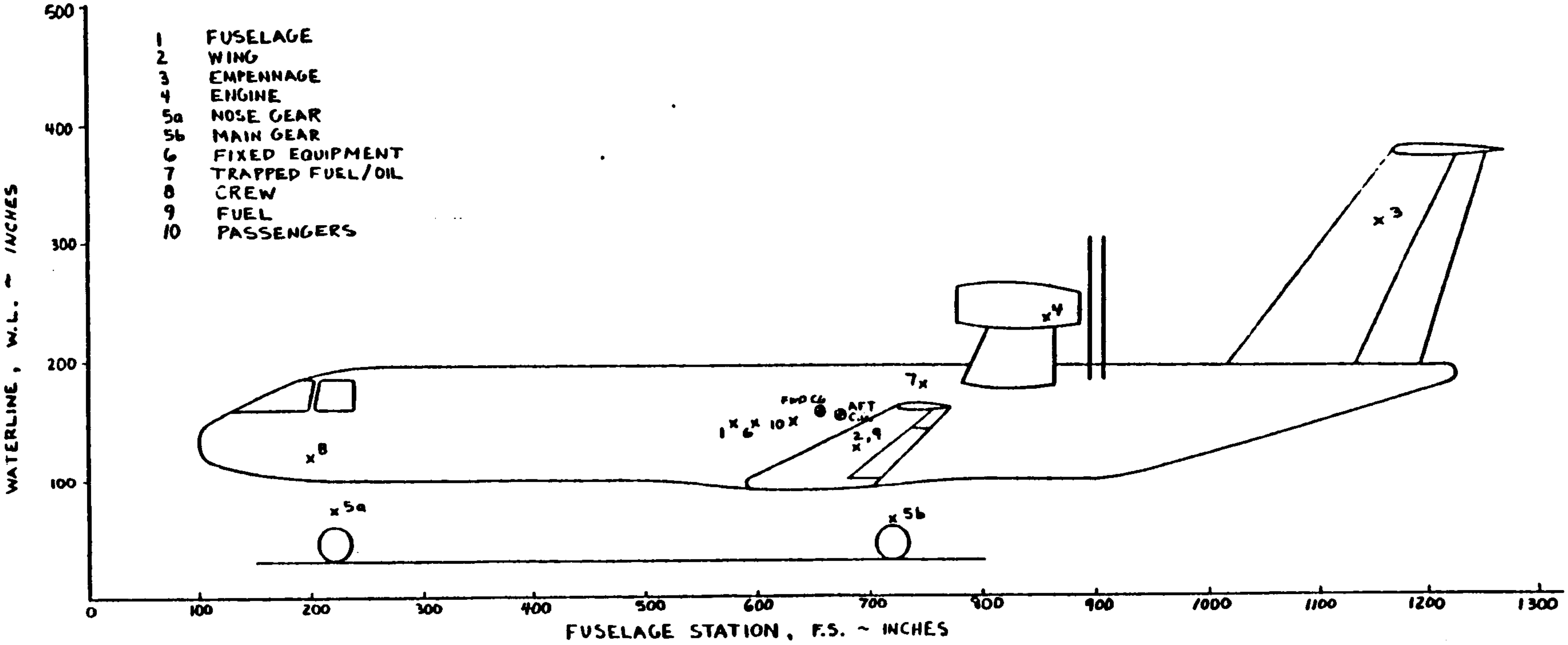
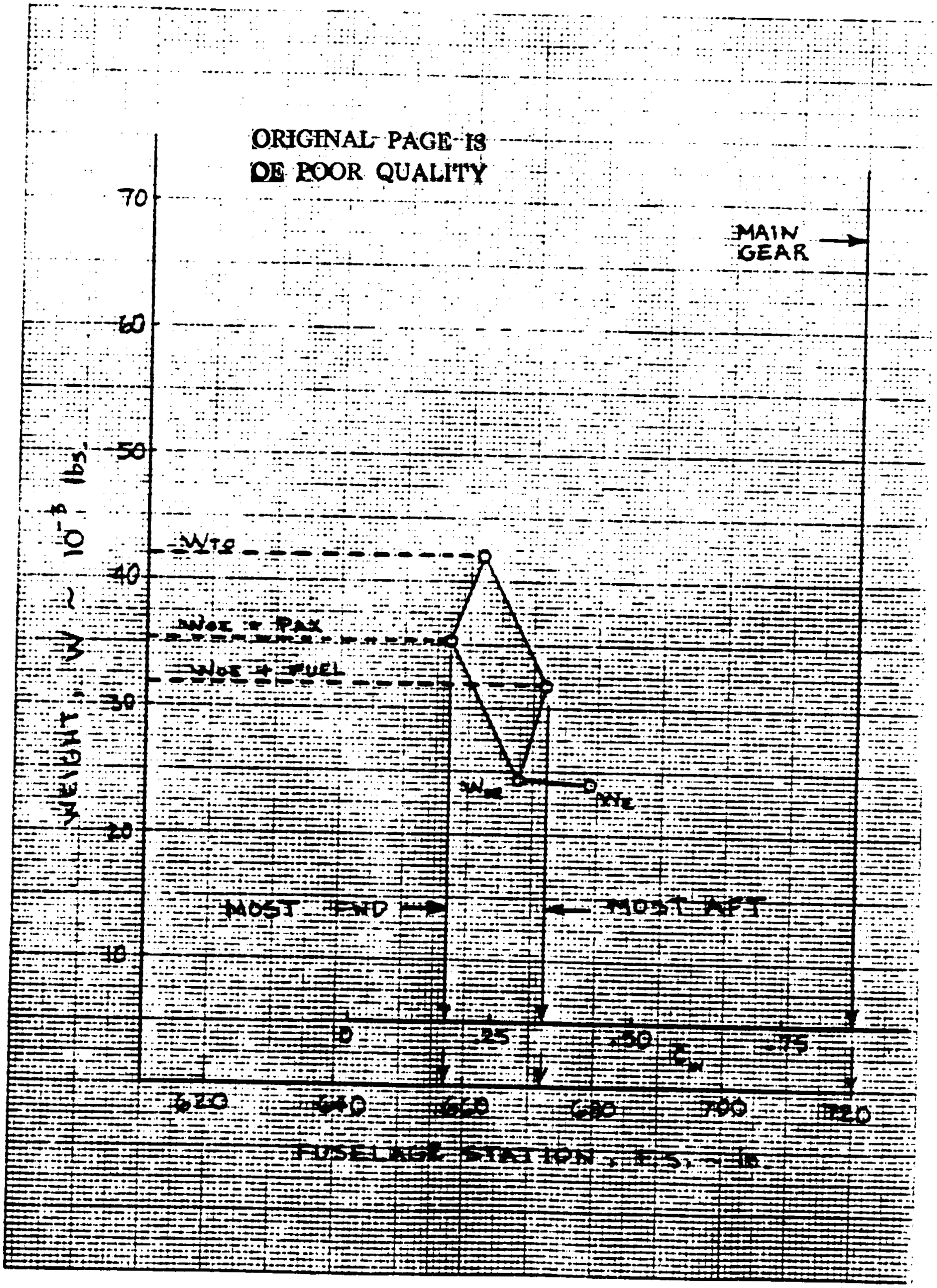


FIGURE 3.4.3 50 PASSENGER GENERAL ARRANGEMENT

Table 3.4.4 50 Passenger Commuter Class I Weight and Balance Calculation

No.	Component	Weight lbs	X_i in	Z_i in
1.	Fuselage	5352	578	148
2.	Wing	4873	687	127
3.	Empennage	1219	1155	340
4.	Engine	4552	855	229
5a.	Nose Gear	373	220	74
5b.	Main Gear	1497	720	64
6.	Fixed Eqpt.	6177	598	148
Empty Weight		$W_E = 24043$		$X_{cg_{We}} = 679$ $Z_{cg_{We}} = 161$
7.	Trapped Fuel and Oil	210	745	178
8.	Crew	615	200	120
Operating Weight Empty:		$W_{OE} = 24868$		$X_{cg_{Woe}} = 668$ $Z_{cg_{Woe}} = 160$
9.	Fuel	6939	687	127
	$W_{OE} + W_F = 31807$			$X_{cg_{Woe+Wf}} = 672$ $Z_{cg_{Woe+Wf}} = 153$
10.	Passengers	10250	630	148
Take-off Weight		$W_{TO} = 42057$		$X_{cg_{Wto}} = 662$ $Z_{cg_{Wto}} = 151$
	$W_{TO} - W_F = 35118$			$X_{cg_{Wto-Wf}} = 657$ $Z_{cg_{Wto-Wf}} = 156$



CALC	M. RUSSELL	10-10	REVISED	DATE	FIGURE 3.4.4 CENTER OF GRAVITY EXCURSION DIAGRAM OF THE 50 PASSENGER MODEL	AE 790
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3.4.9 Stability and Control Analysis

A Class I stability and control analysis was performed using methods of Reference 2, Chapter 11. Table 3.4.5 lists the geometric quantities and stability derivatives necessary to size the empennage from stability and control considerations. Design calculations are located in Appendix J, Section J.7.

3.4.9.1 Longitudinal Stability

From methods in Chapter 11 of Reference 2, the horizontal tail was resized to incorporate a desired static margin of 5 percent. In order to achieve a common horizontal tail with the twin body 100 passenger design, it was necessary to size the 50 passenger horizontal tail to a static margin of 12.9 percent.
Figure J.2 in Appendix J shows that a longitudinal tail area of 102 ft^2 is required. This area will be used in place of the original estimate of Section 3.4.5.

3.4.9.2 Lateral-Directional Stability

From methods in Chapter 11 of Reference 2, the vertical tail area required to hold engine-out flight was critical. The engines were put at a 5 degree cant to lessen the thrust moment arm about the airplane center of gravity. This allowed for a vertical tail area of 170 ft^2. Figure J.3 in Appendix J contains a directional x-plot forthe airplane. It is observed that a 170 ft^2 vertical tail yields $c_{n_\beta} = 0.0958\ rad^{-1}$.

3.4.10 Class I Drag Polars

From methods in Reference 2, Chapter 12, component wetted areas were calculated and listed in Table 3.4.6. The calculations for the wetted areas are given in Appendix J, Section J.8. From the total airplane wetted area and assuming a skin friction coefficient of $C_f = 0.0025$, $C_{D_o} = 0.0169$ was determined.

Table 3.4.7 contains the take-off, cruise and landing drag polars computed during the initial performance sizing. Changes to C_{D_o} for take-off and landing drag polars are given in Appendix J, Section J.8.

Taking natural laminar flow into account should reduce the airplane C_D by at least 10 percent. Assuming $C_{L_{CR}} = 0.3$, $(L/D)_{CR} = 14.1$. During initial take-off weight sizing $(L/D)_{CR}$ was assumed to be 16. It appears that an increase in take-off weight is necessary. From the take-off weight sensitivities given in Table 3.4.3, this change in (L/D) results in an increase in take-off weight of 1889 lbs, or 4.5%. This amount change does not warrant resizing of the airplane, assuming that the 10% reduction in parasite drag is possible.

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Table 3.4.5 Stability and Control Results for the 50 Passenger Commuter

$S = 592\ \text{ft}^2$

$\bar{c} = 7.46\ \text{ft}$ L.E. $\bar{c}_w$ = F.S. 642

$b = 84.3\ \text{ft}$

$S_H = 102\ \text{ft}^2$

$S_V = 170\ \text{ft}^2$

$\Delta\bar{X}_{ac_B} = -0.308$

$\bar{X}_{ac_{WB}} = -0.058$

$\bar{X}_{ac_A} = 0.465$

$\bar{X}_{ac_H} = 6.35$

$C_{L_{\alpha_W}} = 4.72\ \text{rad}^{-1}$

$C_{L_{\alpha_H}} = 3.64\ \text{rad}^{-1}$

$C_{L_{\alpha_V}} = 1.87\ \text{rad}^{-1}$

$C_{n_\beta} = 0.0958\ \text{rad}^{-1}$

$\partial\epsilon/\partial\alpha = 0.325$

$\bar{X}_{cg_{aft}} = 0.335$ F.S. 672

$X_V = 37.2\ \text{ft}$

$X_H = 41.1\ \text{ft}$

*All results calculated from References 5 and 6.

Table 3.4.6 Wetted Area Breakdown

Component	Wetted Area
Wing	1059
Horizontal Tail	207
Vertical Tail	351
Fuselage	2115
Engine Nacelles	180
Engine Pylons	124

Total = 4036 ft^2

$f = 12.1$ $C_f = .0025$

$C_{D_o} = 0.0169$

Table 3.4.7 Drag Polar Comparison

Flight Condition	Initial	Class I
Take-off	$C_D = 0.0634 + 0.0332C_L^2$	$C_D = 0.0354 + 0.0332C_L^2$
Cruise	$C_D = 0.0286 + 0.0312C_L^2$	$C_D = 0.0206 + 0.0312C_L^2$
Landing	$C_D = 0.0784 + 0.0332C_L^2$	$C_D = 0.110 + 0.0332C_L^2$

$(L/D)_{max}$

	Initial	Class I
Take-off	10.9	14.6
Cruise	16.8	19.7
Landing	9.81	8.28

3.5 PRESENTATION OF THE 100 PASSENGER TWIN FUSELAGE CONFIGURATION

Figure 3.5.1 contains the Class I 3-view from the 100 passenger twin body commuter. Table 3.5.1 contains the geometry of the configuration.

3.5.1 Initial Sizing of the 100 Passenger Twin Body

The 100 passenger twin body design is based on joining two optimally designed 50 passenger configurations, in hopes that:

1) high commonality in design and production between the 50 and 100 passenger configurations can be achieved,
2) the weight can be reduced from a conventional passenger configuration,
3) an innovative, futuristic design for the next century can be obtained.

The mission specifications and profile are provided in Table 3.5.2. The initial weight and performance sizing is based on the 50 passenger design and is listed in Table 3.5.3.

3.5.2 Fuselage and Cockpit Layouts

The 100 passenger twin fuselage design has the same cockpit and fuselage cross section as the rest of the commuter family with one exception: the right-hand side fuselage cockpit will be stripped of equipment and used as additional seating or for observation. The cockpit design and fuselage cross section are contained in Appendix A. The lengths of the fuselage and cabin are given in Table 3.5.1. The design methodology followed the steps in References 2 and 3.

3.5.2 Fuselage and Cockpit Layouts

The commuter family will be powered by two advanced turboprop engines. The 100 passenger twin body requires the use of the 13,500 shp turboprops. Appendix B contains the engine data used.

3.5.4 Wing and Flap Design

Table 3.5.1 presents the geometry of the wing and flaps. The wing planform and flaps are the same as that used on the 50 passenger airplane. A center wing joining the two fuselages and connected to the outboard wings was added. The center wing had the following characteristics:

Area, S = 400 ft^2

Thickness Ratio, $t/c = 0.13$

Dihedral Angle and incidence angle, = $i = 0$ deg

The flaps were sized to a $C_{L_{max_L}} = 3.0$. This required

Table 3.5.1 Geometric Characteristics of the Twin Fuselage 100 Passenger Commuter Airplane

	Wing	Horizontal Tail	Vertical Tail
Area	923 ft^2	354 ft^2	140 ft^2
Span	118 ft	55.8 ft	15.4 ft
MGC	8.33 ft	4.68	9.4 ft
MGC L.E. : F.S.	51.8 ft	99.9	91.7 ft
Aspect Ratio	15	5.0	1.7
Sweep Angle	15.0 deg	25.0 deg (L.E.)	40.0 deg (L.E.)
Taper Ratio	0.40	0.50	0.50
Thickness Ratio	0.13 root 0.10 tip	0.12 root 0.10 tip	0.13 root 0.12 tip
Airfoil:	All airfoils are Natural Laminar Flow airfoils.		
Dihedral Angle	7.0 deg	0 deg	0 deg
Incidence Angle	0 deg	0 deg	0 deg
Aileron chord ratio	0.25	Elevator chord ratio = 0.36 (full span)	Rudder chord ratio = 0.34 (full span)
Aileron span ratio	0.750 - 0.97		
Flap chord ratio	0.25		
Flap span ratio	0.10 - 0.738		

	Fuselage	Cabin Interior	Overall
Length	94.6 ft	45.0 ft	104 ft
Maximum Height	8.05 ft	6.30 ft	28.0 ft
Maximum Width	8.05 ft	7.60 ft	118 ft

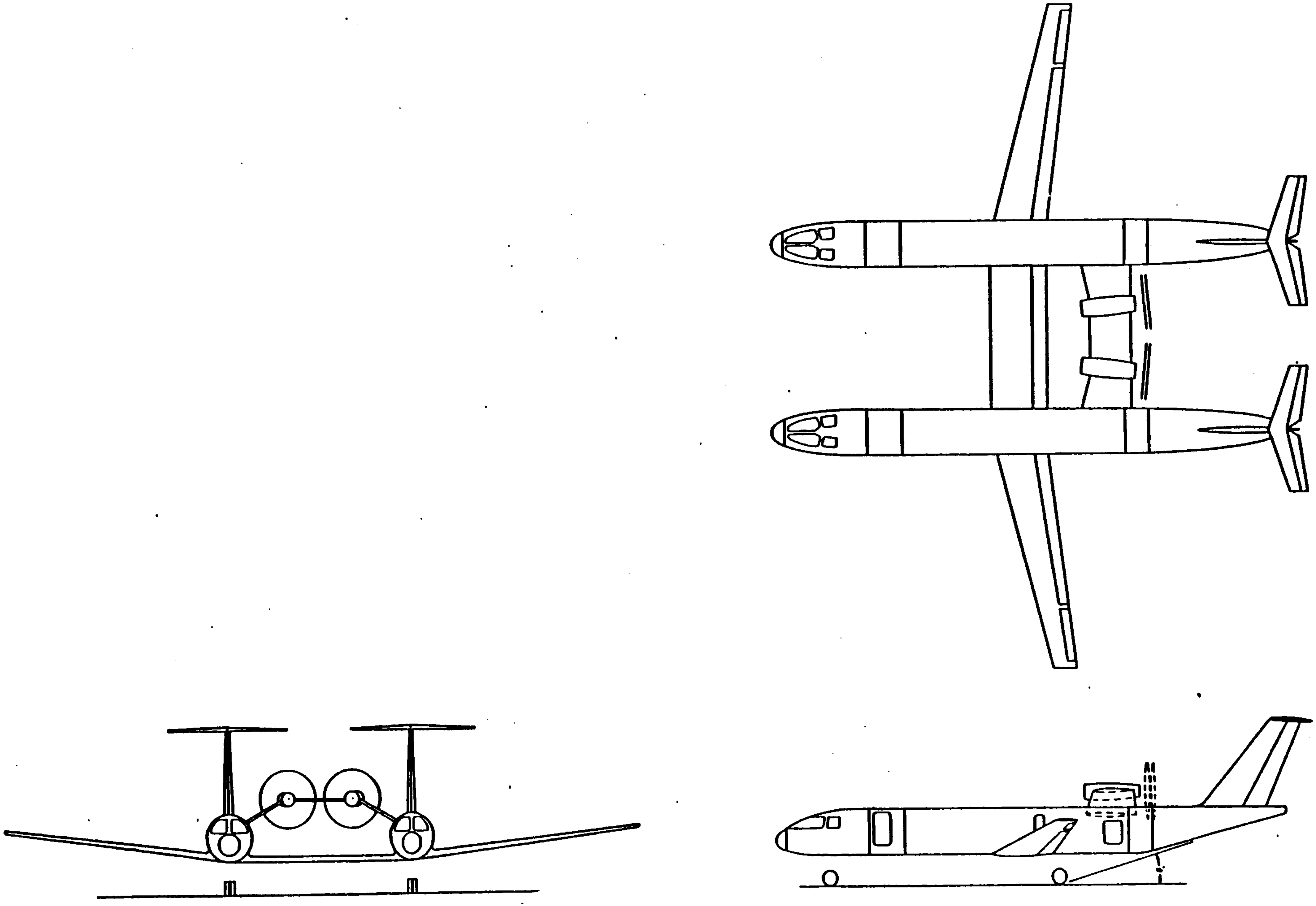

FIGURE 3.5.1 3-VIEW OF THE 100 TWIN-BODY MODEL

Table 3.5.2 Mission Specification for the Twin Body 100 Passenger Advanced Technology Commuter Airplane

PAYLOAD:	100 passengers at 175 lbs each with 30 lbs of baggage per passenger, carry-on luggage capability is required
CREW:	2 pilots and 2 flight attendants at 175 lbs each with 30 lbs of baggage each
RANGE:	1500 nm with max payload with 25% fuel reserves
ALTITUDE:	30,000 ft at the design range
CRUISE SPEED:	MACH = .70
CLIMB:	climb rate of 3000 fpm
TAKE-OFF AND LANDING:	3500 ft balanced field length
POWERPLANTS:	advanced turboprops
PRESSURIZATION:	5000 ft cabin at 30000 ft
CERTIFICATION BASE:	FAR 25

MISSION PROFILE:

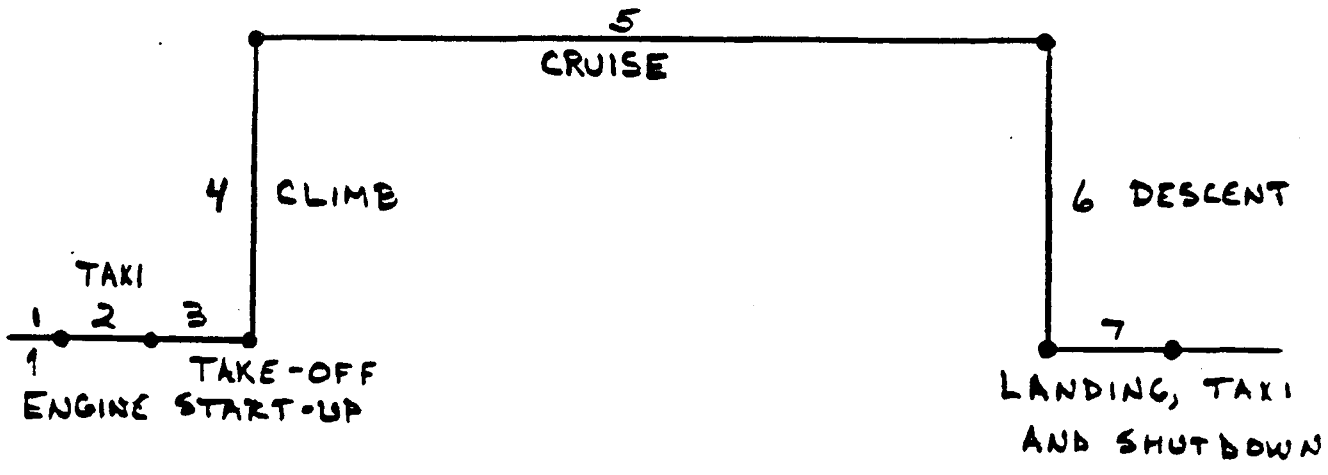

Table 3.5.3 Initial Sizing Parameters for the Twin Body 100 Passenger Commuter

Weights:

Take-off Weight W_{TO} = 80,716 lbs

Operating Weight Empty W_{OE} = 46,338 lbs

Payload Weight W_{PL} = 20,500 lbs

Crew Weight W_{CREW} = 615 lbs

Mission Fuel Weight W_F = 13,878 lbs

Wing Area $S = 923\ ft^2$ $M_{FF} = .828$

Aspect Ratio $A = 15.0$

Take-off Power P_{TO} = 22,000 shp

Required Lift Coefficients:

Clean $C_{L_{max}} = 1.5$

Take-off $C_{L_{max_{TO}}} = 2.0$

Landing $C_{L_{max_L}} = 3.0$

Take-off Weight Sensitivities*:

$\partial W_{TO} / \partial c_p$ = 39,784 lb/lb/hp/hr

$\partial W_{TO} / \partial \eta_p$ = -18,722 lbs

$\partial W_{TO} / \partial (L/D)$ = -994.6 lbs

$\partial W_{TO} / \partial R$ = 15.1 lbs

*assumed to be the same as the 50 passenger commuter

the use of Fowler flaps on the 50 passenger airplane. The center wing section has been designed to include full span flaps if needed.

Section 2.4.4 gives the details on the 50 passenger wing planform and flap design used for this configuration.

3.5.5 Design of the Empennage

Table 3.5.1 lists the empennage geometry for the 100 passenger twin body. Initially, the areas obtained by the V-bar method for the 50 passenger design (see Section 2.4.5) were doubled:

$$S_V = 260 \text{ ft}^2$$

$$S_H = 260 \text{ ft}^2$$

However, the empennage was redesigned from stability and control considerations of both the 100 passenger twin body and 50 passenger designs in Sections 2.4.9 and 3.5.9.

3.5.6 Control Surface Sizing

3.5.6.1 Lateral-Directional Controls

Table 3.5.1 presents the aileron geometry used. It is the same as designed for the 50 passenger design. Spoilers may be required in order to produce the extra roll-control required for a twin-fuselage design.

3.5.6.2 Longitudinal Controls

The elevators are the same as those for the 50 passenger design; the geometry is summarized in Table 3.5.1.

3.5.7 Landing Gear Design

From Chapter 9, Reference 2, it was determined that a 30 X 9 inch tire could be used on every airplane of the commuter family. A preliminary retraction scheme for the main gear is shown in Appendix D. The gear placement is the same as that for the 50 passenger airplane. The wheelbase for the 100 passenger twin body has been estimated to be 50 ft. From Airport Engineering by Ashford and Wright, the following conclusions are made:

1) This design can operate out of any airline airport.
2) This design will not be able to operate out of general aviation airports. General and basic transport general aviation airports have taxiway widths between 40 - 60 feet.

3.5.8 Class I Weight and Balance Calculations

A preliminary weight and balance of the 100 passenger twin body was determined by using methods in Reference 2, Chapter 10. Component weights and center of gravity locations are contained in Table 3.5.4. A general arrangement drawing is provided by Figure 2.4.2. The weight-center of gravity excursion diagram is contained in Figure 3.5.3. The 100 passenger twin body has a 22 inch excursion range which corresponds to 0.22 $\bar{c}_w$.

3.5.9 Stability and Control Analysis

A Class I stability and control analysis was performed using methods of Reference 2, Chapter 11. Table 3.5.5 lists all the geometric quantities and stability derivatives necessary to size the empennage from stability and control considerations. Appendix N (pages 8-22) provides the detailed calculations.

3.5.9.1 Longitudinal Stability

From methods in Chapter 11 of Reference 2, the horizontal tail was resized to best match that of the 50 passenger design while still maintaining an inherently stable static margin. Figure N.2 in Appendix N presents the longitudinal x-plot for the airplane. Since only 102 ft^2 of horizontal tail area was required by the 50 passenger design, a horizontal boom has been proposed to connect the horizontal tail planforms (see Figure 3.5.1). This provides a horizontal tail area of 303 ft^2 and allows the design of an inherently stable static margin of 7.5 percent.

3.5.9.2 Lateral-Directional Stability

From methods in Chapter 11 of Reference 2, the vertical tail area required to hold engine-out flight was not critical. Figure 3.5.5 provides the directional x-plot. The 100 passenger twin body only requires 230 ft^2 of vertical tail area; however, the 50 passenger design required 170 ft^2 due to engine-out requirements. The 100 passenger twin body will use two 140 ft^2 vertical tails. From this the following results:

$S_V = 280 \text{ ft}^2$

$C_{n_\beta} = 0.098 \text{ rad}^{-1}$

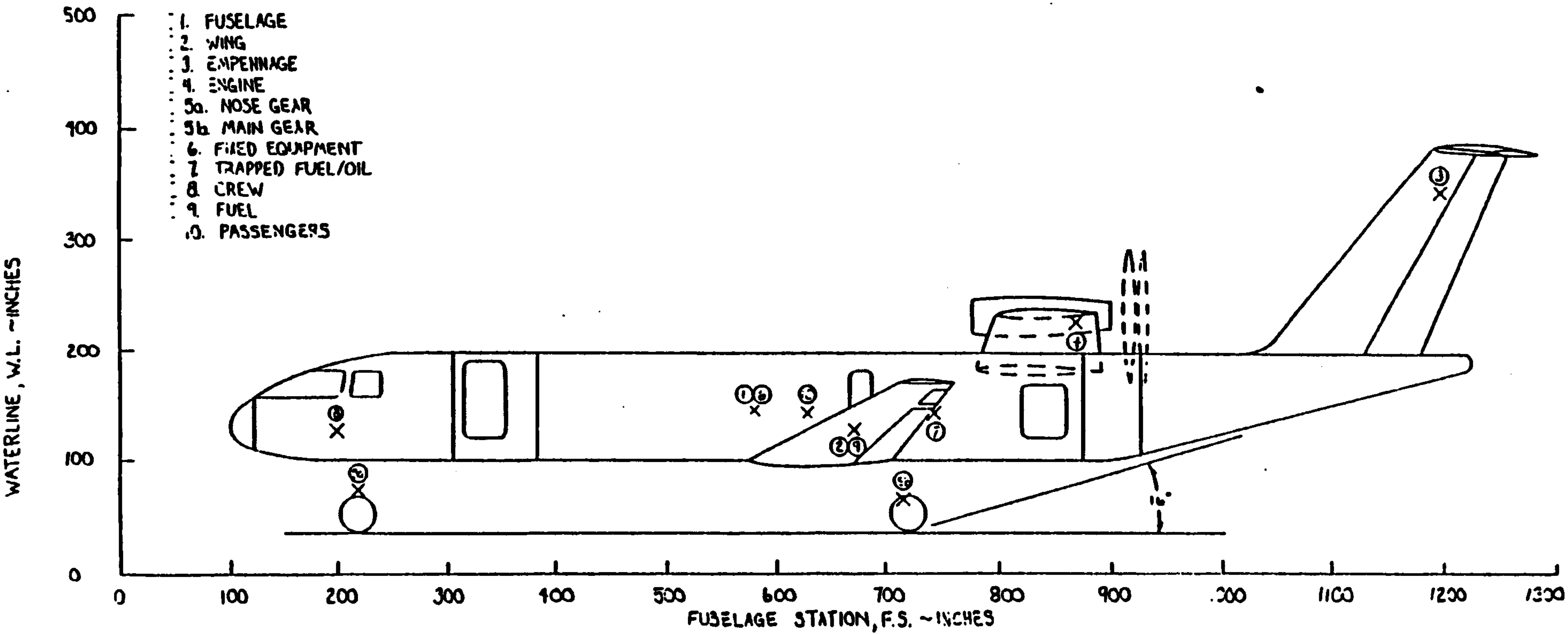

FIGURE 3.5.2 100 TWIN-BODY GENERAL ARRANGEMENT

Table 3.5.4 Twin Body 100 Passenger Commuter Class I Weight and Balance Calculation

No.	Component	Weight lbs	X_i in	Z_i in
1.	Fuselage	10704	578	148
2.	Wing	7597	672	127
3.	Empennage	2438	1204	340
4.	Engine	8470	870	222
5a.	Nose Gear	746	220	74
5b.	Main Gear	2994	720	64
6.	Fixed Eqpt.	12354	578	148
Empty Weight		$W_E = 45303$		$X_{cg_{We}} = 686$
				$Z_{cg_{We}} = 161$
7.	Trapped Fuel and Oil	420	745	178
8.	Crew	615	200	120
Operating Weight Empty:		$W_{OE} = 46338$		$X_{cg_{Woe}} = 680$
				$Z_{cg_{Woe}} = 160$
9.	Fuel	13878	672	127
	$W_{OE} + W_F = 60216$			$X_{cg_{Woe+Wf}} = 678$
10.	Passengers	20500	630	148
Take-off Weight		$W_{TO} = 80716$		$X_{cg_{Wto}} = 666$
				$Z_{cg_{Wto}} = 151$
	$W_{TO} - W_F = 66838$			$X_{cg_{Wto-Wf}} = 665$

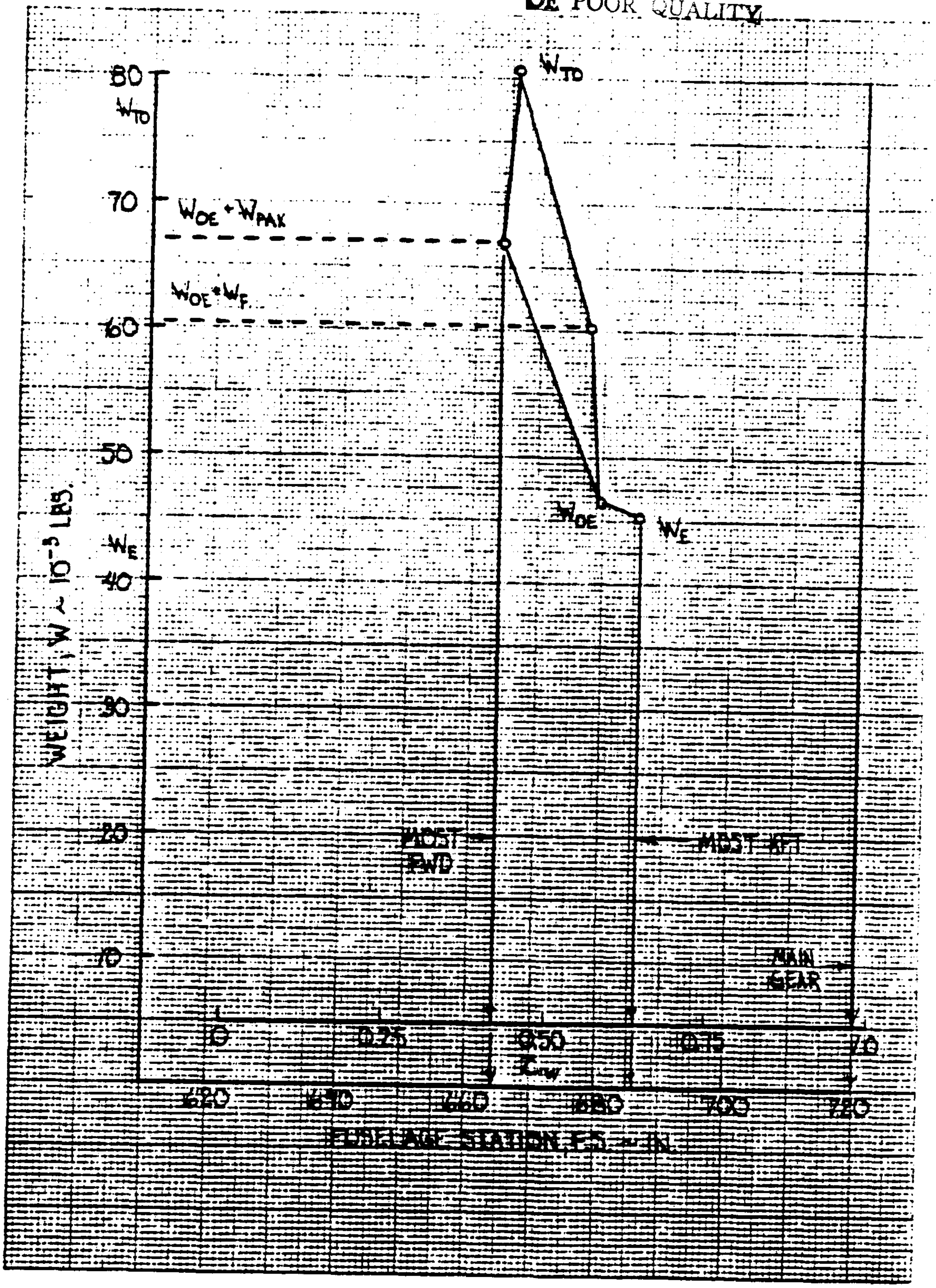

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Table 3.5.5 Stability and Control Results for the Twin Body 100 Passenger Commuter

$S = 923 \text{ ft}^2$

$\bar{c} = 8.33 \text{ ft}$ LE $\bar{c}_w$ = F.S. 622

$b = 118 \text{ ft}$

$S_H = 354 \text{ ft}^2$

$S_V = 280 \text{ ft}^2$

$\Delta\bar{x}_{ac_B} = -0.390$

$\bar{x}_{ac_{WB}} = -0.140$

$\bar{x}_{ac_A} = 0.622$ F.S. 684

$\bar{x}_{ac_H} = 6.50$ $\bar{x}_{ac_{H_e}} = 1.71$

$C_{L_{\alpha_W}} = 5.20 \text{ rad}^{-1}$

$C_{L_{\alpha_H}} = 3.69 \text{ rad}^{-1}$ $C_{L_{\alpha_{H_e}}} = 4.13 \text{ rad}^{-1}$

$C_{L_{\alpha_V}} = 2.14 \text{ rad}^{-1}$

$C_{n_\beta} = 0.098 \text{ rad}^{-1}$

$d\epsilon/d\alpha = 0.344$

$\bar{x}_{cg_{aft}} = 0.580$ F.S. 680

$x_V = 41.3 \text{ ft}$

*All results calculated from References 5 and 6.

3.5.10 Class I Drag Polars

From methods in Reference 2, Chapter 12, component wetted areas were calculated and listed in Table 3.5.6. The calculations for the wetted areas are given in Appendix N, Section N.8. From the total airplane wetted area and assuming a skin friction coefficient of $c_f = 0.0025$, $C_{D_o} = 0.0184$ was determined. Table 3.5.7 contains take-off, cruise, and landing drag polars which result. Changes to C_{D_o} for take-off and landing drag polars are given Appendix N, Section N.8.

Assuming $C_{L_{CR}} = 0.3$, $(L/D)_{CR} = 14.4$. This decrease in $(L/D)_{CR}$ from that of the 50 passenger design was anticipated due to the large increase in wetted area in key places: fuselage, engine pylons, and center wing surfaces. However, if 10 percent laminar flow is assumed as in the 50 passenger design, $(L/D)_{CR} = 15.8$. This corresponds to the design goal of $(L/D)_{CR} = 16$. Detailed calculations are provided in Appendix N (pages 23-27).

Table 3.5.6 Wetted Area Breakdown

Component	Wetted Area
Wing	1042
Horizontal Tail	625
Vertical Tail	567
Fuselage	4270
Engine Nacelles	393
Pylons	315
Total =	7212 ft^2

$f = 17.0$ $C_f = .0025$

$C_{D_o} = 0.0184$

Table 3.5.7 Twin Body Drag Polars

Flight Condition	Class I Drag Polar	$(L/D)_{max}$
Take-off	$C_D = 0.0334 + 0.0265C_L^2$	16.8
Cruise	$C_D = 0.0186 + 0.0250C_L^2$	23.2
Landing	$C_D = 0.1084 + 0.0265C_L^2$	9.32

3.6 PRELIMINARY DESIGN OF THE 75 PASSENGER BASELINE CONFIGURATION

The purpose of this chapter is to present the preliminary design of the 75 passenger regional transport. Figure 3.6.1 shows the Class I three-view of the NASA-100. Table 3.6.1 presents the geometric parameters for the NASA-100.

3.6.1 INITIAL WEIGHT AND PERFORMANCE SIZING FOR THE 75 PASSENGER BASELINE CONFIGURATION

3.6.1.1 INITIAL WEIGHT SIZING

Initial weight sizing was conducted using a method in Reference 1. The following assumptions were made for the airplane:

1) $(L/D)_{cr} = 16$

2) $c_p = 0.4$ lbs/hp/hr

The above assumptions and the mission specifications, given in Table 3.6.2, yielded the airplane weights and sensitivities in Table 3.6.3. Appendix K, section K.2, contains output from XEWTOG, a computerized weight sizing method developed at the University of Kansas.

3.6.1.2 INITIAL PERFORMANCE SIZING

XPRFRM, a computer program developed at the University of Kansas, was used to determine the required take-off power, P_{TO}, and wing area, S, that meet the performance criteria given in Table 3.6.2. XPRFRM follows the method of Reference 1. Maximum lift coefficients and wing aspect ratio are also determined. Figure 3.6.2 shows the required power loading, wing loading combinations that satisfy the performance criteria. From Figure 3.6.2, it is determined that cruise speed and landing field length requirements are critical for this airplane. The results of the performance sizing effort are listed in Table 3.6.2. Appendix K, section K.3, details the computer output of XPRFRM.

3.6.2 FUSELAGE AND COCKPIT LAYOUTS

The fuselage and cockpit layouts were determined using the methods of Chapter 4 in Ref. 2 and Chapter 2 in Ref. 3.

The 75 passenger transport has the same flight deck layout and fuselage cross-section as the rest of the

TABLE 3.6.1--TABLE OF GEOMETRY FOR THE 75 PASSENGER COMMUTER.

	Wing	Horizontal Tail	Vertical Tail
Area, S (ft^2)	1178	134	363
Span, b (ft)	119	26.7	22.5
MGC, $\bar{c}$ (ft)	10.5	5.4	16.4
MGC LE: F.S.			
Aspect ratio, A	12	5.3	1.4
Sweep angle, (deg)	13 (c/4)	22 (c/4)	42 (c/4)
Taper ratio,	0.4	0.35	0.6
Thickness ratio, t/c	0.13	0.13	0.13
Airfoil	NLF	NLF	NLF
Dihedral, (deg)	7	0	90
Incidence, i (deg)	0	Variable	0
Spoiler:		Elevator:	Rudder:
Chord ratio	0.14	0.39/0.45	0.35
Span location	0.43/0.70		
Flaps:			
Chord ratio:	0.25		
Span ratio:	0.07/1.00		

	Fuselage	Cabin Interior	Overall
Length, l (ft)	108	67.5	121
Maximum width, (ft)	8.05	7.60	119
Maximum heighth, (ft)	14.0	6.30	36.9

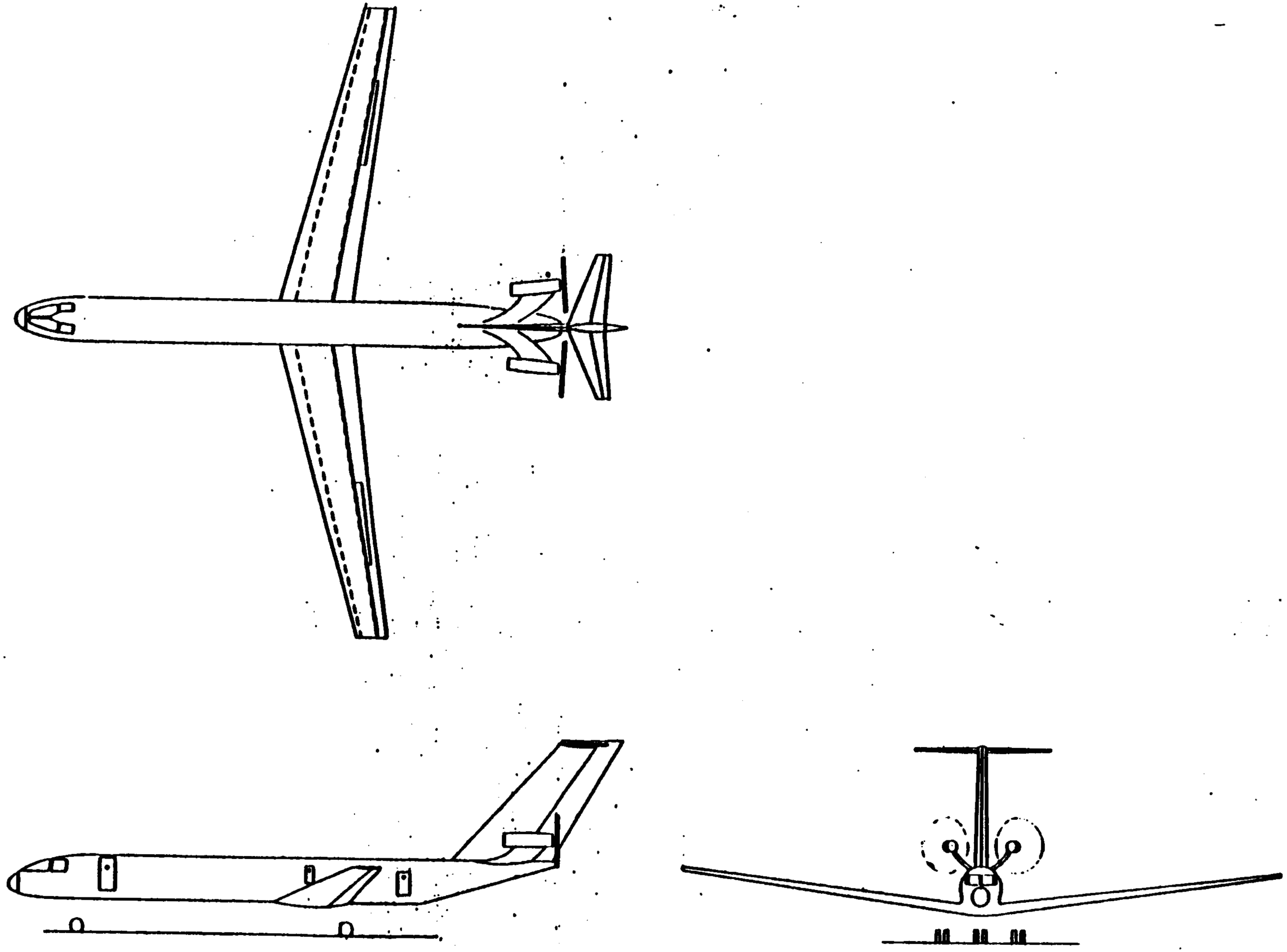

FIGURE 3.6.1 3-VIEW OF THE 75 PASSENGER MODEL

TABLE 3.6.2--MISSION SPECIFICATION FOR A 75 PASSENGER ADVANCED TECHNOLOGY COMMUTER AIRPLANE

PAYLOAD:	75 passengers at 175 lbs each with 30 lbs of baggage per passenger, carry-on luggage capability is required
CREW:	2 pilots and 2 flight attendants at 175 lbs each with 30 lbs of baggage each
RANGE:	1500 nm with max payload with 25% fuel reserves
ALTITUDE:	30,000 ft at the design range
CRUISE SPEED:	MACH = .70
CLIMB:	climb rate of 3000 fpm
TAKE-OFF AND LANDING:	3500 ft balanced field length
POWERPLANTS:	advanced turboprops
PRESSURIZATION:	5000 ft cabin at 30,000 ft
CERTIFICATION BASE:	FAR 25

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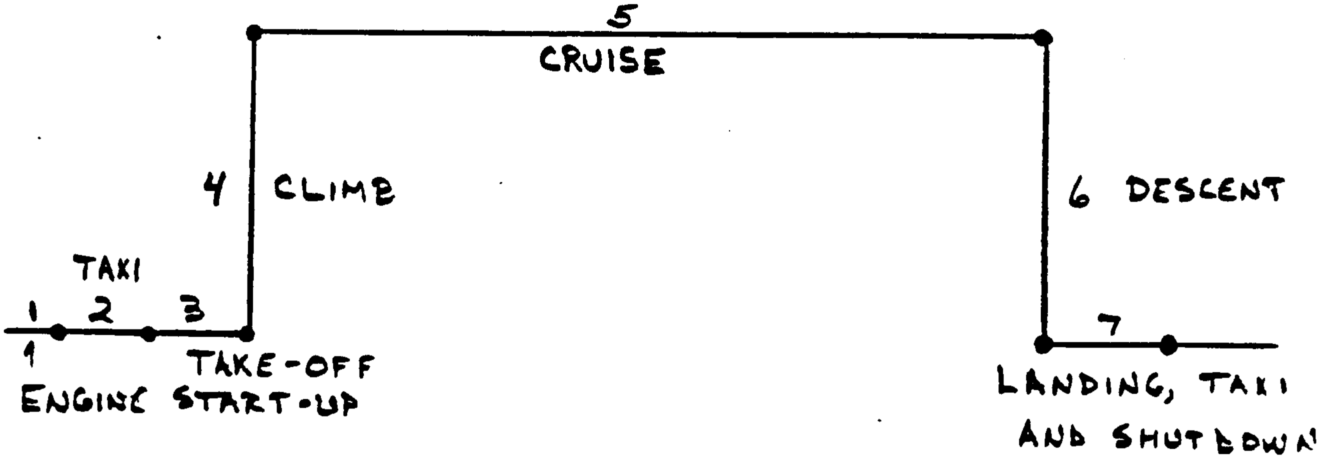


TABLE 3.6.3--INITIAL SIZING PARAMETERS FOR THE 75 PASSENGER COMMUTER.

Weights: Take-off weight - $W_{TO} = 82,491$ lbs.
Empty weight - $W_{E} = 48,175$ lbs.
Payload weight - $W_{PL} = 15,375$ lbs.
Mission fuel weight - $W_{F} = 17,898$ lbs.
Crew weight - $W_{CREW} = 820$ lbs.

Wing area: $S = 1178\ \text{ft}^2$.

Wing Aspect ratio: $A = 12$.

Take-off power: $P_{TO} = 19,640$ lbs.

Required lift coefficients: Clean, $C_{L_{max}} = 1.40$.
Take-off, $C_{L_{max}} = 1.80$.
Landing, $C_{L_{max}} = 3.00$.

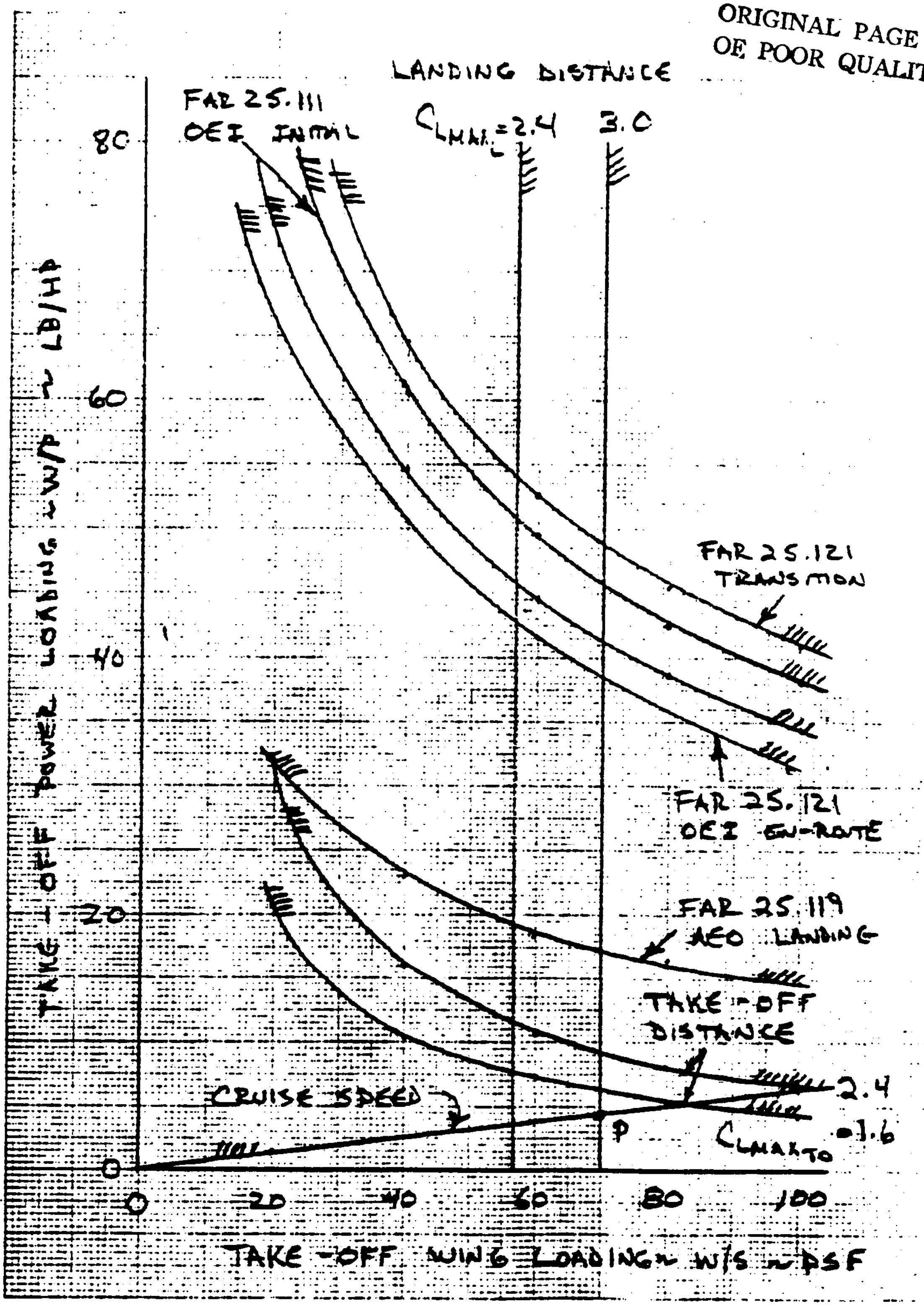
Take-off weight sensitivities:

SFC - $\partial W_{TO}/\partial c_p = 143,189$ lb/lb/lb/hr

Propeller efficiency- $\partial W_{TO}/\partial \eta_p = -67,383$ lbs

Lift-to-drag ratio - $\partial W_{TO}/\partial (L/D) = -3,579$ lbs

Range - $\partial W_{TO}/\partial R = 38.2$ lb/nm



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commuter family. Appendix A contains the fuselage cross-section and cockpit layout design. Table 3.6.1 gives the main dimensions of the fuselage.

3.6.3 ENGINE SELECTION

The engines were selected using the methods of Chapter 5 in Ref. 2. Two advanced turbo-props were chosen at a power rating of 13,500 shp per engine. The engine data is given in Appendix B.

3.6.4 WING AND FLAP DESIGN

Table 3.6.1 presents the geometry of the wing and flaps. Parameters such as leading edge sweep and wing thickness were decided by the selection of an NLF airfoil. Appendix C contains the airfoil cross section and airfoil parameters. Wing parameters were selected using the method of Chapter 6 in Ref. 2.

The flaps were sized to a $C_{L_{max_L}} = 3.0$. This required the use of Fowler flaps. The sizing methods used are contained in Chapter 7 of Ref. 2. The design calculations are in Appendix K, section K.4.

3.6.5 DESIGN OF THE EMPENNAGE

Table 3.6.1 shows the empennage for the 75 passenger airplane. Initially, the V-bar method in chapter 8 of Ref. 2 was used to size the empennage. The design calculations are in Appendix K, section K.5. The initial tail areas that resulted are listed below:

$$S_H = 242 \text{ ft}^2$$

$$S_V = 363 \text{ ft}^2$$

After the stability and control calculations of Section 3.6.9 were completed, the empennage was resized. These considerations are discussed in section 3.6.9.

3.6.6 CONTROL SURFACE SIZING

3.6.6.1 LATERAL - DIRECTIONAL CONTROLS

Since full span flaps were required for landing, spoilers were used in place of ailerons. The spoiler geometry is contained in Table 3.6.1. This geometry was determined from Chapter 8 of Ref. 2.

The rudder was also sized with the method of Chapter 8 in Ref. 2. The rudder geometry is given in Table 3.6.1.

3.6.6.2 LONGITUDINAL CONTROLS

The elevators were sized using the methods in Chapter 8 of Ref. 2. Geometric parameters for the elevators are presented in Table 3.6.1.

3.6.7 LANDING GEAR DESIGN

From Chapter 9 of Ref. 2, it was determined that a 30" x 9" tire could be utilized for the nose and main landing gear on every airplane of the commuter family. A preliminary retraction scheme for the main gear is shown in Appendix D. The gear placement was dictated by weight and balance calculations shown in section 3.6.8.

Both the longitudinal and the lateral tip-over criterion were satisfied. Appendix K, section K.6, contains the lateral tip-over calculations.

3.6.8 CLASS I WEIGHT AND BALANCE CALCULATIONS

The weight and balance for the NASA-100 was done after calculating the Class I component weights for the airplane. The component weights were calculated using average weight fractions for the commuter category of airplanes. Appendix F contains the Class I weight fractions for the commuter family. The preliminary weight and balance was then determined using the methods of Chapter 10 in Ref. 2.

The weight breakdown and the center of gravity locations are presented in Table 3.6.4. The center of gravity travel was contained to a range of 30 inches. This travel is 0.21 $\bar{c}_w$. Figure 3.6.4 diagrams the center of gravity excursion for the 75 passenger airplane. Fig. 3.6.4 locates the component cg's on the airplane three-view.

3.6.9 STABILITY AND CONTROL RESULTS

Chapter 11 of Ref. 2 outlines the methods used in the preliminary stability and control calculations. Ref. 5 and Ref. 6 were used as supplements for these calculations. Table 3.6.5 contains geometric quantities and stability derivatives necessary to size the empennage for inherent stability. Design calculations are located in Appendix K, section K.7.

3.6.9.1 LONGITUDINAL STABILITY

From methods in Chapter 11 of Ref. 2, the horizontal tail was resized to incorporate a desired static margin of 5%.

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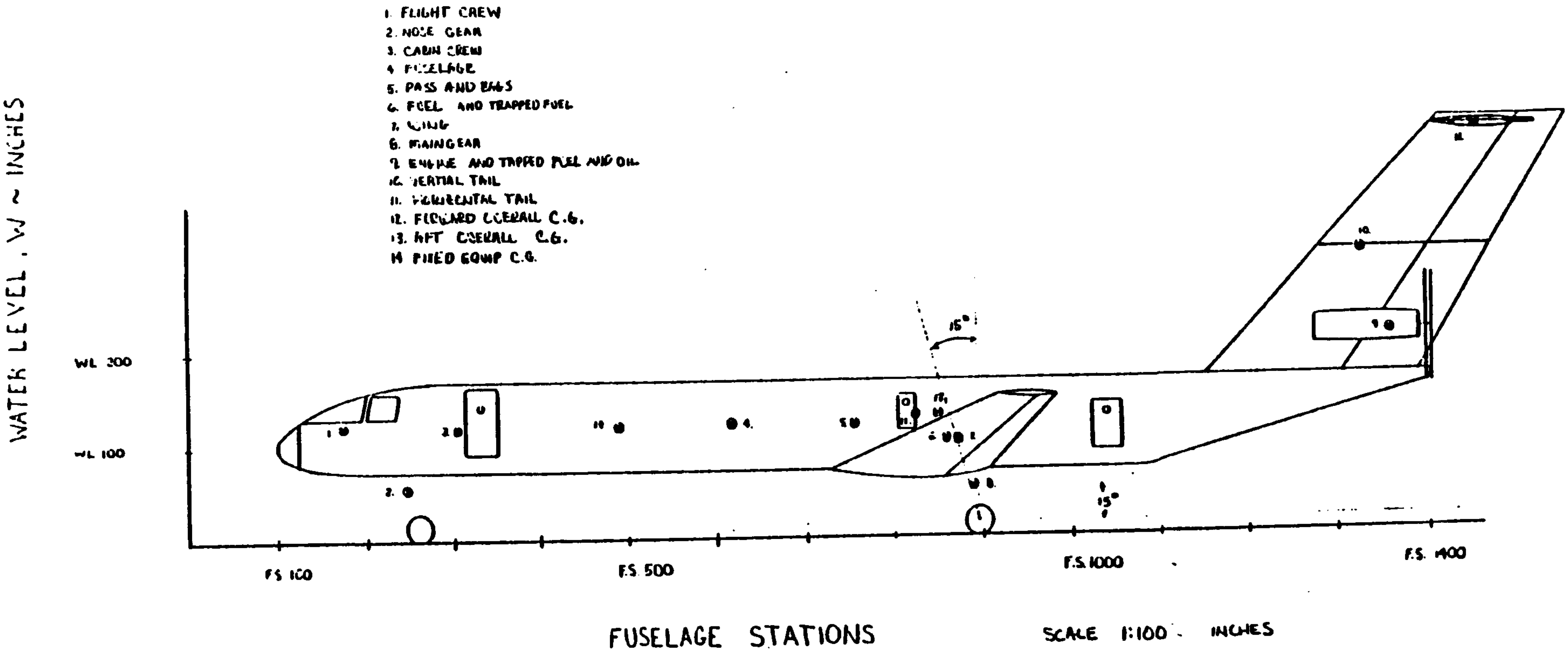

FIGURE 3.6.3 75 PASSENGER GENERAL ARRANGEMENT

TABLE 3.6.4--CLASS I WEIGHT AND BALANCE CALCULATION FOR THE 75 PASSENGER COMMUTER.

No.	Type of Component	W_i lbs	x_i in	$W_i x_i$ inlbs	z_i in	$W_i z_i$ inlbs
1.	Fuselage	10,311	6.2	6.372×10^4	1.25	1.114×10^4
2.	Wing	9,404	8.7	8.182×10^4	1.06	0.858×10^4
3.	Empennage	2,392	13.8	3.300×10^4	3.77	0.883×10^4
4.	Engine	10,288	13.6	1.399×10^5	2.17	2.233×10^4
5.	Landing gear					
	a. Nose gear	709	2.5	0.172×10^4	0.58	0.031×10^4
	b. Main gear	2,837	8.8	2.497×10^4	0.58	0.165×10^4
6.	Fixed eqpt.	12,044	4.9	5.883×10^4	1.25	1.505×10^4

Empty weight: W_E = 47,986 lbs

x_{cgWe} = 846 in

z_{cgWe} = 136 in

No.	Type of Component	W_i lbs	x_i in	$W_i x_i$ inlbs	z_i in	$W_i z_i$ inlbs
7.	Trapped fuel and oil	411	10.6	4.377×10^3	1.49	0.613×10^3
8.	Crew	820	2.5	1.976×10^3	1.26	0.513×10^3

Operating empty weight: W_{OE} = 49,218 lbs.

x_{cgWoe} = 838 in

z_{cgWoe} = 136 in

No.	Type of Component	W_i lbs	x_i in	$W_i x_i$ inlbs	z_i in	$W_i z_i$ inlbs
9.	Fuel	17,898	8.6	1.546×10^5	1.04	1.861×10^4
10.	Passengers	15,375	7.5	1.159×10^5	1.25	1.922×10^4

Take-off weight: W_{TO} = 82,491 lbs.

x_{cgWto} = 818 in

z_{cgWto} = 127 in

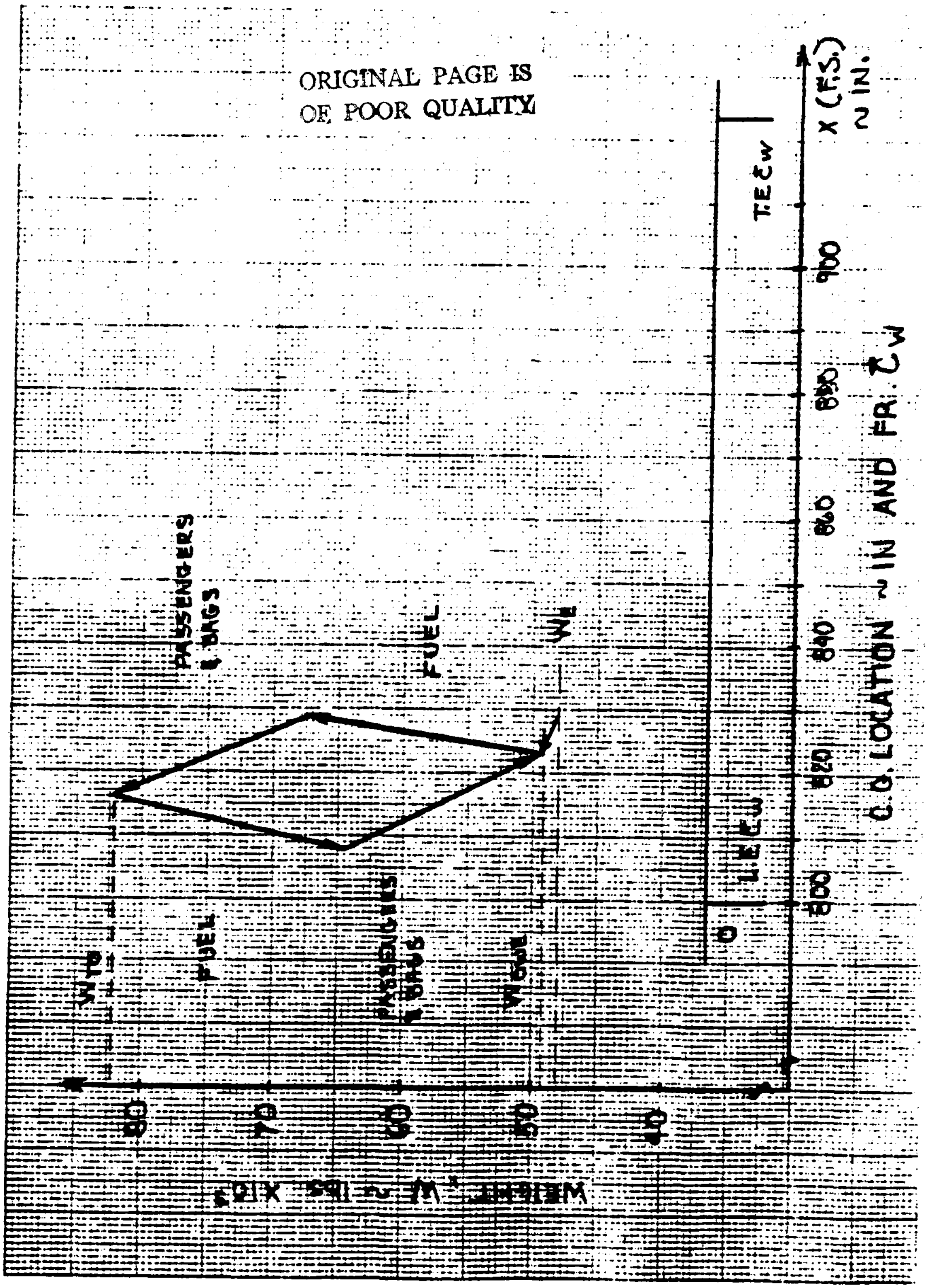

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FIGURE 3.6.4 CENTER OF GRAVITY EXCURSION DIAGRAM OF THE 75 PASSENGER MODEL

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TABLE 3.6.5--STABILITY AND CONTROL RESULTS FOR THE 75 PASSENGER COMMUTER.

$S = 1178\ ft^2$; $\bar{c} = 10.5$ ft ; $b = 119$ ft.

$\Delta \bar{x}_{ac_B} = -0.13$

$\bar{x}_{ac_{WB}} = 0.12$

$\bar{x}_{ac_A} = 0.428$

$\bar{x}_{ac_H} = 4.87$

$C_{L_{\alpha_W}} = 4.71\ rad^{-1}$

$C_{L_{\alpha_H}} = 3.51\ rad^{-1}$

$C_{L_{\alpha_V}} = 1.43\ rad^{-1}$

$C_{n_\beta} = 0.0573\ rad^{-1}$

$\partial\epsilon/\partial\alpha = 0.185$

$\bar{x}_{cg_{aft}} = 0.29$

$x_V = 36.7$ ft.

Appendix K, Figure K.2 presents the longitudinal x-plot for the 75 passenger airplane. From this plot, it is seen that a tail area of 134 ft^2 is required.

3.6.9.2 LATERAL - DIRECTIONAL STABILITY

From the method of Chapter 11 in Ref. 2, the vertical tail area required to hold engine-out flight was found to be critical. The engines were set at a five degree cant to lessen the thrust moment arm about the cg. The directional x-plot is given in Appendix K, Figure K.3. From this plot, it can be seen that a vertical tail area of 363 ft^2 yields a $C_{n_\beta} = 0.0010\ deg^{-1}$.

3.6.10 CLASS I DRAG POLARS

The Class I drag polars were calculated from the procedure of Chapter 12 in Ref. 2. The wetted areas of the airplane components were calculated as presented in Table 3.6.6 and Appendix K, section K.8. From the total airplane wetted area and assuming a skin friction coefficient of 0.0025, C_{D_o} for the airplane was calculated.

Table 3.6.7 contains the take-off, cruise, and landing drag polars computed during the initial performance sizing. These drag polars are compared to the drag polars computed from wetted area considerations. These Class I drag polars more accurately represent the airplane. Changes to C_{D_o} for take-off and landing conditions are given in Appendix K, section K.8.

The clean zero-lift drag coefficient at low speed was determined as:

$$C_{D_o} = 0.0124$$

The drag polars for take-off, landing, and cruise were then calculated as shown in Appendix K.

Assuming a $C_{L_{cr}} = 0.3$, the final drag polars yield:

$$(L/D)_{cr} = 12.4$$

During initial take-off weight sizing, $(L/D)_{cr}$ was assumed to be 16.

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TABLE 3.6.6--WETTED AREA BREAKDOWN.

	Component	S_{wet} (ft^2)
1.	Wing	2212
2.	Horizontal tail	277
3.	Vertical tail	750
4.	Fuselage	2463
5.	Engines	1248
6.	Engine pylons	25
	Total	5975

From Figure 3.2.1 of Reference 1, assuming a $c_f = .0025$:

$$f = 14.6 \ ft^2$$
$$C_{D_o} = f/S_{REF} = 14.6/1178 = 0.0124$$

TABLE 3.6.7--DRAG POLAR COMPARISON.

Preliminary Results	Drag Polar	$(L/D)_{max}$
1. Clean	$0.0208 + 0.0312C_L^2$	19.6
2. Take-off, gear down	$0.0358 + 0.0332C_L^2$	14.5
3. Landing, gear down	$0.0958 + 0.0332C_L^2$	8.9

$(L/D)_{cruise \ at \ C_L = 0.3} = 12.7$

Class I Results	Drag Polar	$(L/D)_{max}$
1. Clean	$0.0214 + 0.0312C_L^2$	19.4
2. Take-off, gear down	$0.0474 + 0.0332C_L^2$	12.6
3. Landing, gear down	$0.1074 + 0.0332C_L^2$	8.4

$(L/D)_{cruise \ at \ C_L = 0.3} = 12.4$

The sensitivities to take-off weight given in Table 3.6.3 show that:

$$\partial W_{TO}/\partial(L/D) = -3,579 \text{ lbs}$$

For the baseline configuration, this translates into:

$$(L/D)_{cr} = 12.4 - 16 = - 3.6$$

$$W_{TO} = \Delta(L/D)_{cr}\ \partial W_{TO}/\partial(L/D) = 12,884 \text{ lbs.}$$

Since the take-off weight is 82,491 lbs, the decrease in lift-to-drag ratio causes a 16% increase in take-off weight. According to Ref. 2, this percentage change in take-off weight indicates that the airplane needs to be resized with the initial weight sizing methods of Ref. 1.

3.7 PRELIMINARY DESIGN OF THE 100 PASSENGER BASELINE CONFIGURATION

The purpose of this chapter is to present the preliminary design of the NASA-100 regional transport. Figure 3.7.1 shows the Class I three-view of the NASA-100. Table 3.7.1 presents the geometric parameters for the NASA-100.

3.7.1 INITIAL WEIGHT AND PERFORMANCE SIZING FOR THE 100 PASSENGER BASELINE CONFIGURATION

3.7.1.1 INITIAL WEIGHT SIZING

Initial weight sizing was conducted using a method in Reference 1. The following assumptions were made for the airplane:

1) $(L/D)_{cr} = 16$

2) $c_p = 0.4$ lbs/hp/hr

The above assumptions and the mission specifications, given in Table 3.7.2, yielded the airplane weights and sensitivities in Table 3.7.3. Appendix L, section L.2, contains output from XEWTOG, a computerized weight sizing method developed at the University of Kansas.

3.7.1.2 INITIAL PERFORMANCE SIZING

XPRFRM, a computer program developed at the University of Kansas, was used to determine the required take-off power, P_{TO}, and wing area, S, that meet the performance criteria given in Table 3.7.2. XPRFRM follows the method of Reference 1. Maximum lift coefficients and wing aspect ratio are also determined. Figure 3.7.2 shows the required power loading, wing loading combinations that satisfy the performance criteria. From Figure 3.7.2, it is determined that cruise speed and landing field length requirements are critical for this airplane. The results of the performance sizing effort are listed in Table 3.7.2. Appendix L, section L.3, details the computer output of XPRFRM.

3.7.2 FUSELAGE AND COCKPIT LAYOUTS

The fuselage and cockpit layouts were determined using the methods of Chapter 4 in Ref. 2 and Chapter 2 in Ref. 3.

The 100 passenger transport has the same flight deck layout and fuselage cross-section as the rest of the

TABLE 3.7.1--TABLE OF GEOMETRY FOR THE 100 PASSENGER COMMUTER.

	Wing	Horizontal Tail	Vertical Tail
Area, S (ft^2)	1604	155	300
Span, b (ft)	139	28.7	20.6
MGC, $\bar{c}$ (ft)	11.6	5.4	15.0
MGC LE: F.S.	925	1675	1530
Aspect ratio, A	12	5.3	1.4
Sweep angle, (deg)	15 (LE)	22 (c/4)	42 (c/4)
Taper ratio,	0.4	0.35	0.6
Thickness ratio, t/c	0.13	0.13	0.13
Airfoil	NLF	NLF	NLF
Dihedral, (deg)	7	0	90
Incidence, i (deg)	0	Variable	0
Spoiler:		Elevator:	Rudder:
Chord ratio	0.23	0.36	0.34
Span location	0.4/0.6		
Hinge line	0.70c		
Aileron:			
Chord ratio:	0.30		
Span ratio:	0.76/1.00		
Flaps:			
Chord ratio:	0.30		
Span ratio:	0.06/0.76		

	Fuselage	Overall
Length, l (ft)	126	137.5
Maximum width, (ft)	8.05	139
Maximum heighth, (ft)	8.05	35.4

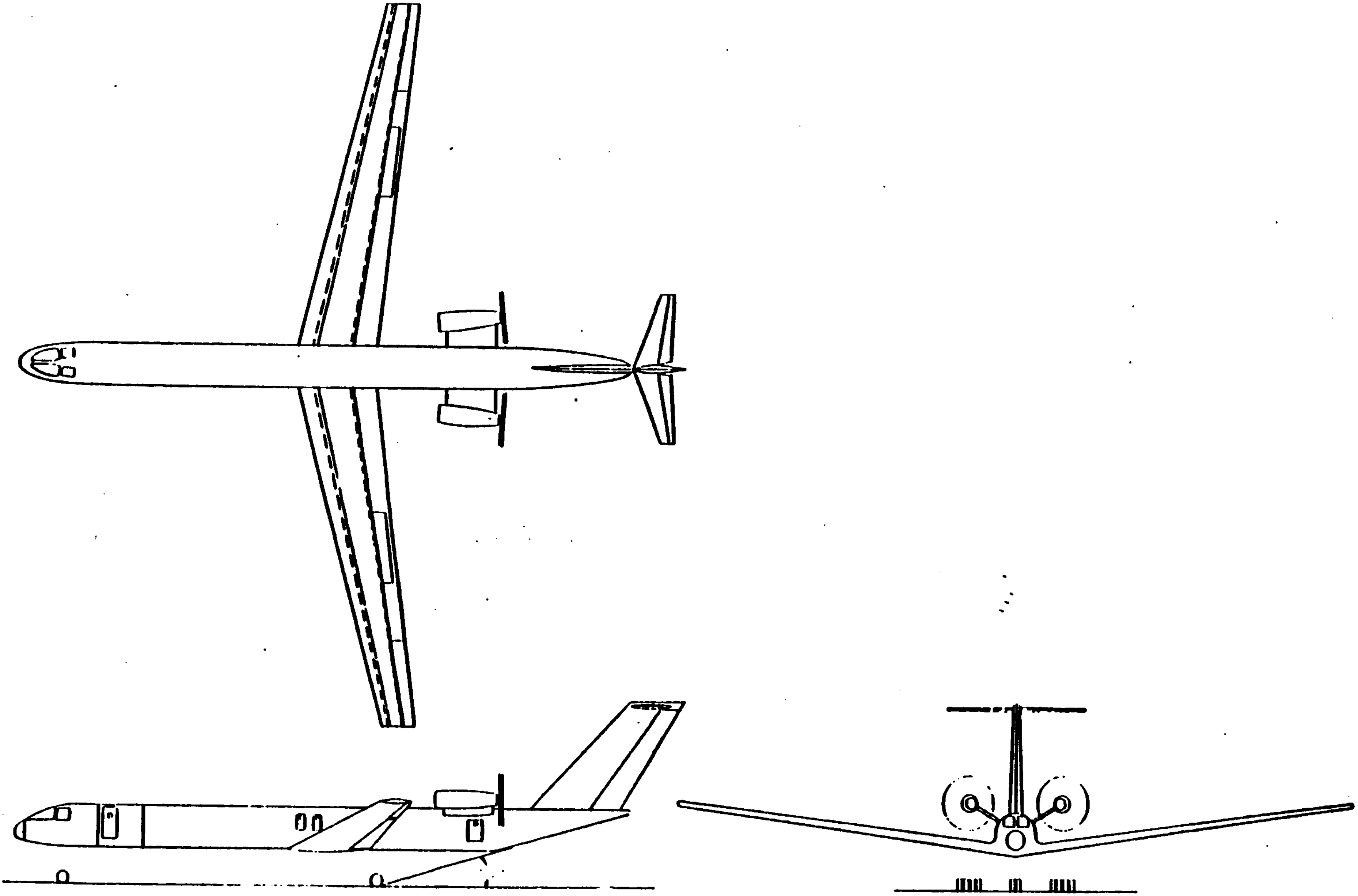
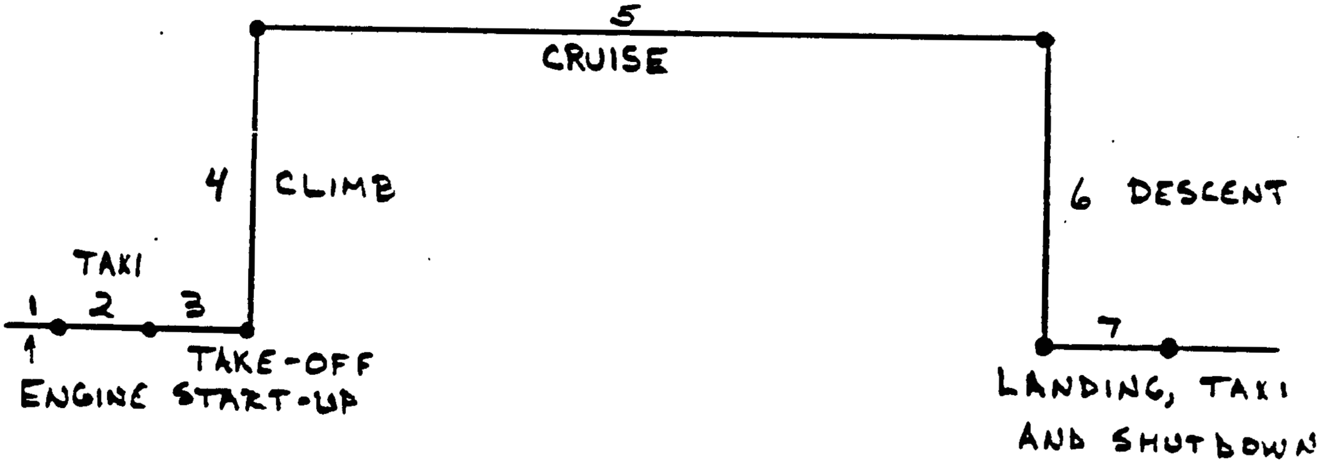


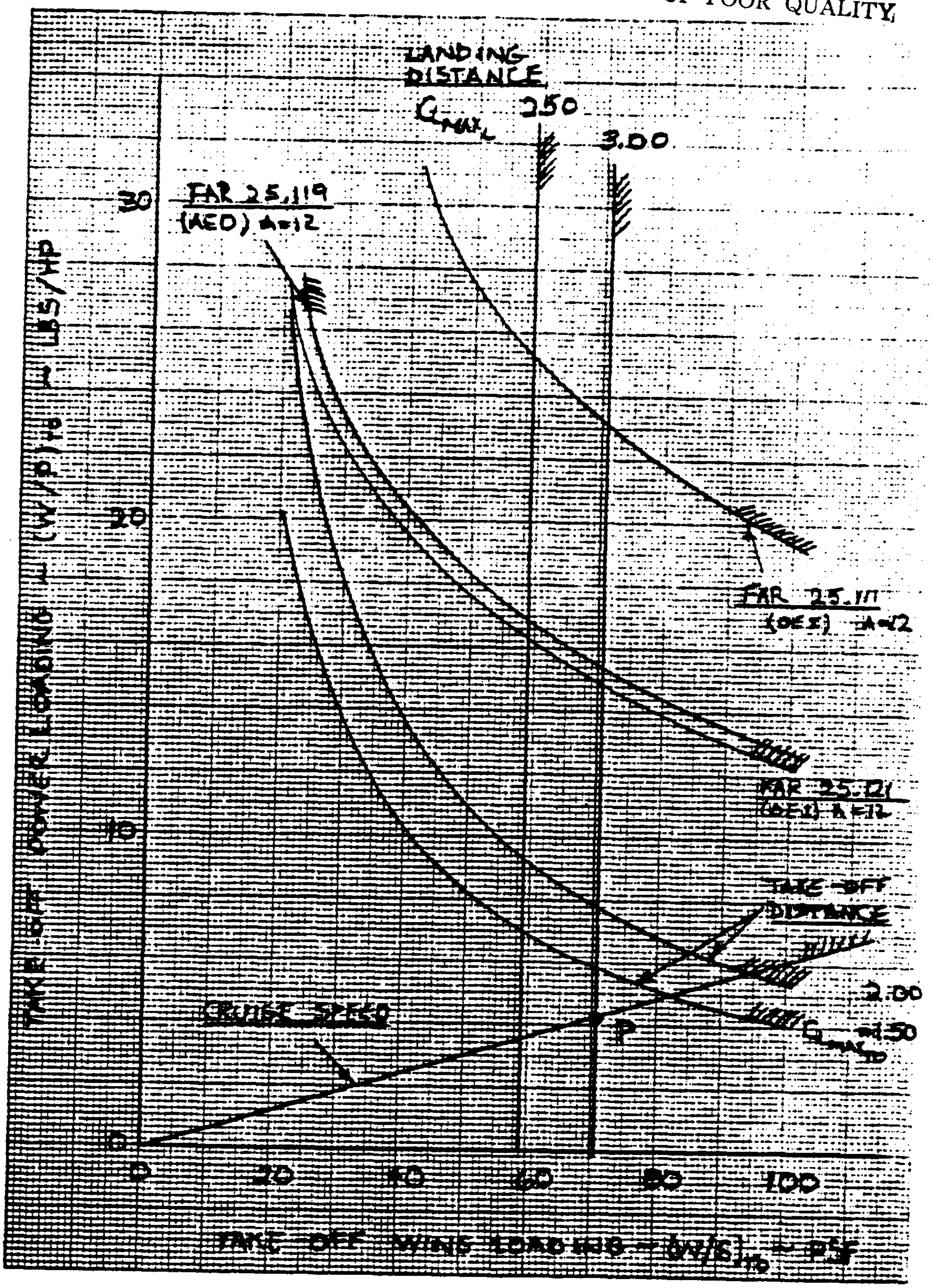
FIGURE 3.7.1 3-VIEW OF THE 100 PASSENGER MODEL

TABLE 3.7.2--MISSION SPECIFICATION FOR A 100 PASSENGER ADVANCED TECHNOLOGY COMMUTER AIRPLANE

PAYLOAD:	100 passengers at 175 lbs each with 30 lbs of baggage per passenger, carry-on luggage capability is required
CREW:	2 pilots and 2 flight attendants at 175 lbs each with 30 lbs of baggage each
RANGE:	1500 nm with max payload with 25% fuel reserves
ALTITUDE:	30,000 ft at the design range
CRUISE SPEED:	MACH = .70
CLIMB:	climb rate of 3000 fpm
TAKE-OFF AND LANDING:	3500 ft balanced field length
POWERPLANTS:	advanced turboprops
PRESSURIZATION:	5000 ft cabin at 30,000 ft
CERTIFICATION BASE:	FAR 25

MISSION PROFILE:





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CALC	9-16-86				**FIGURE 3.7.2** **PERFORMANCE MATCHING OF THE 100 PASSENGER MODEL**
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TABLE 3.7.3--INITIAL SIZING PARAMETERS FOR THE 100 PASSENGER COMMUTER.

Weights: Take-off weight - W_{TO} = 112,288 lbs.
Operating weight empty - W_{OWE} = 67,422 lbs.
Empty weight - W_E = 66,041 lbs.
Payload weight - W_{PL} = 20,500 lbs.
Mission fuel weight - W_F = 24,366 lbs.
Crew weight - W_{CREW} = 820 lbs.

Wing area: $S = 1604 \text{ ft}^2$.

Wing Aspect ratio: $A = 12$.

Take-off power: P_{TO} = 26,750 lbs.

Required lift coefficients: Clean, $C_{L_{max}} = 1.32$.
Take-off, $C_{L_{max}} = 1.80$.
Landing, $C_{L_{max}} = 3.00$.

Take-off weight sensitivities:
Payload weight - $\partial W_{TO}/\partial W_{PL} = 5.9$

Empty weight - $\partial W_{TO}/\partial W_E = 1.6$

SFC - $\partial W_{TO}/\partial c_p$ = 202,659 lb/lb/lb/hr

Propeller efficiency- $\partial W_{TO}/\partial \eta_p$ = -95,369 lbs

Lift-to-drag ratio - $\partial W_{TO}/\partial (L/D)$ = -5,067 lbs

Range - $\partial W_{TO}/\partial R$ = 54.0 lb/nm

commuter family. Appendix A contains the fuselage cross-section and cockpit layout design. Table 3.7.1 gives the main dimensions of the fuselage.

3.7.3 ENGINE SELECTION

The engines were selected using the methods of Chapter 5 in Ref. 2. Two advanced turbo-props were chosen at a power rating of 13,500 shp per engine. The required total shaft horsepower was 26,740 hp. The engine data is given in Appendix B.

3.7.4 WING AND FLAP DESIGN

Table 3.7.1 presents the geometry of the wing and flaps. Parameters such as leading edge sweep and wing thickness were decided by the selection of an NLF airfoil. Appendix C contains the airfoil cross section and airfoil parameters. Wing parameters were selected using the method of Chapter 6 in Ref. 2.

The flaps were sized to a $C_{L_{max_L}} = 3.0$. This required the use of Fowler flaps. The sizing methods used are contained in Chapter 7 of Ref. 2. The design calculations are in Appendix L, section L.4.

3.7.5 DESIGN OF THE EMPENNAGE

Table 3.7.1 shows the empennage for the 100 passenger airplane. Initially, the V-bar method in chapter 8 of Ref. 2 was used to size the empennage. The design calculations are in Appendix L, section L.5. The initial tail areas that resulted are listed below:

$$S_H = 347 \text{ ft}^2$$

$$S_V = 378 \text{ ft}^2$$

After the stability and control calculations of Section 3.7.9 were completed, the empennage was resized to:

$$S_H = 155 \text{ ft}^2$$

$$S_V = 303 \text{ ft}^2$$

These considerations are discussed in section 3.7.9.

3.7.6 CONTROL SURFACE SIZING

3.7.6.1 LATERAL - DIRECTIONAL CONTROLS

Both ailerons and spoilers were used on the 100 passenger regional transport. The geometry of both is contained in Table 3.7.1. This geometry was determined from Chapter 8 of Ref. 2.

The rudder was also sized with the method of Chapter 8 in Ref. 2. The rudder geometry is given in Table 3.7.1.

3.7.6.2 LONGITUDINAL CONTROLS

The elevators for the NASA-100 were sized according to the procedure in Chapter 8 of Ref. 2. Geometric parameters for the elevators are presented in Table 3.7.1.

3.7.7 LANDING GEAR DESIGN

From Chapter 9 of Ref. 2, it was determined that a 30" x 9" tire could be utilized for the nose and main landing gear on every airplane of the commuter family. A preliminary retraction scheme for the main gear is shown in Appendix D. The gear placement was dictated by weight and balance calculations shown in section 3.7.8.

Both the longitudinal and the lateral tip-over criterion were satisfied. Appendix L, section L.6, contains the lateral tip-over calculations.

3.7.8 CLASS I WEIGHT AND BALANCE CALCULATIONS

The weight and balance for the NASA-100 was done after calculating the Class I component weights for the airplane. The component weights were calculated using average weight fractions for the commuter category of airplanes. Appendix F contains the Class I weight fractions for the commuter family. The preliminary weight and balance was then determined using the methods of Chapter 10 in Ref. 2.

The weight breakdown and the center of gravity locations are presented in Table 3.7.4. The center of gravity travel was contained to a range of 36 inches. This travel is approximately 2% of the overall length, or $0.26\ \bar{c}_w$. Figure 3.7.4 diagrams the various center of gravity locations at different airplane weights. Fig. 3.7.4 locates the component cg's on the NASA-100 three-view.

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3.7.9 STABILITY AND CONTROL RESULTS

Chapter 11 of Ref. 2 outlines the methods used in the preliminary stability and control calculations. Ref. 5 and

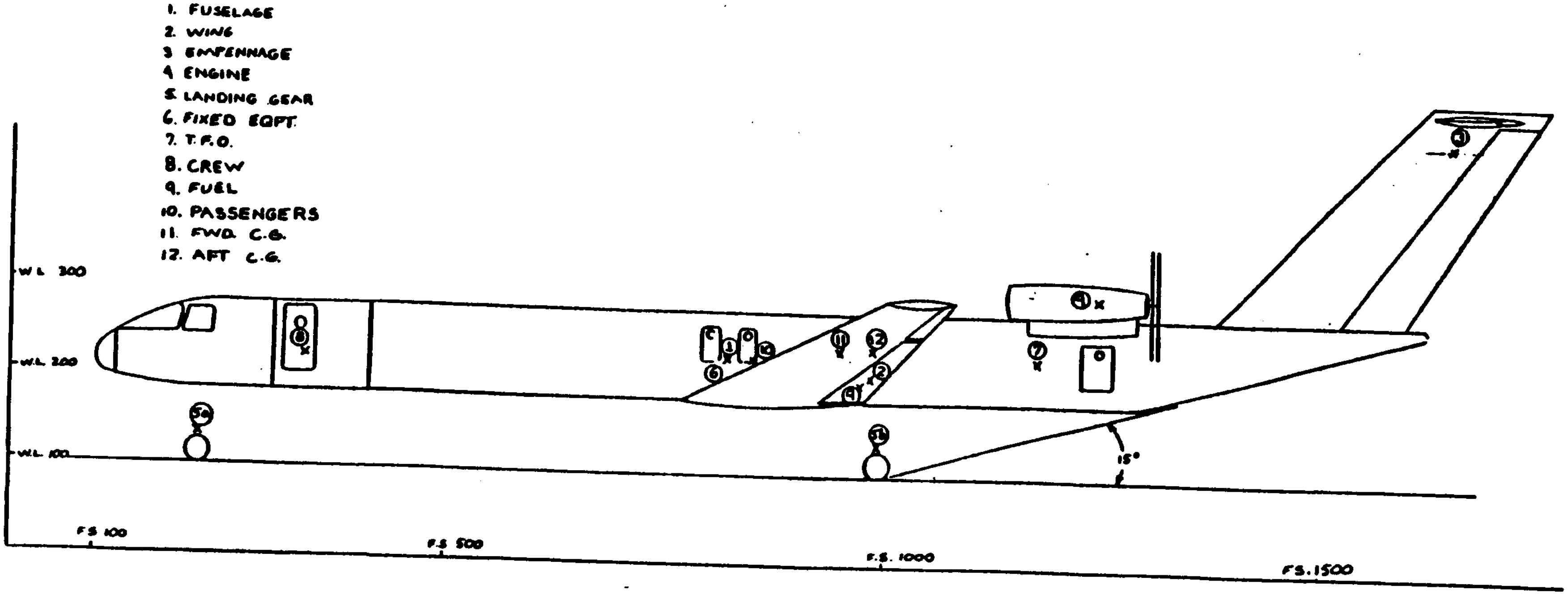

FIGURE 3.7.3 100 PASSENGER GENERAL ARRANGEMENT

TABLE 3.7.4--CLASS I WEIGHT AND BALANCE CALCULATION FOR THE 100 PASSENGER COMMUTER.

No.	Type of Component	W_i lbs	x_i in	W_ix_i inlbs	z_i in	W_iz_i inlbs
1.	Fuselage	14,260	829	1.182×10^7	230	3.280×10^6
2.	Wing	13,043	985	1.285×10^7	212	2.765×10^6
3.	Empennage	3,256	1640	0.534×10^7	488	1.589×10^6
4.	Engine	14,260	1230	1.754×10^7	305	4.349×10^6
5.	Landing gear					
	a. Nose gear	966	246	0.024×10^7	130	0.126×10^6
	b. Main gear	3,862	1005	0.388×10^7	130	0.502×10^6
6.	Fixed eqpt.	16,394	829	1.359×10^7	230	3.771×10^6

Empty weight: W_E = 66,042 lbs x_{cgWe} = 988 in

z_{cgWe} = 248 in

No.	Type of Component	W_i lbs	x_i in	W_ix_i inlbs	z_i in	W_iz_i inlbs
7.	Trapped fuel and oil	561	1175	6.592×10^5	230	1.290×10^5
8.	Crew	820	340	2.788×10^5	230	1.886×10^5

Operating empty weight: W_{OE} = 67,422 lbs. x_{cgWoe} = 982 in

z_{cgWoe} = 248 in

No.	Type of Component	W_i lbs	x_i in	W_ix_i inlbs	z_i in	W_iz_i inlbs
9.	Fuel	24,366	975	2.376×10^7	208	5.068×10^6
10.	Passengers	20,500	855	1.753×10^7	230	4.715×10^6

Take-off weight: W_{TO} = 112,288 lbs. x_{cgWto} = 957 in

z_{cgWto} = 236 in

x_{cg} (Woe + Pass) = 952 in. x_{cg} (Woe + Fuel) = 980 in.

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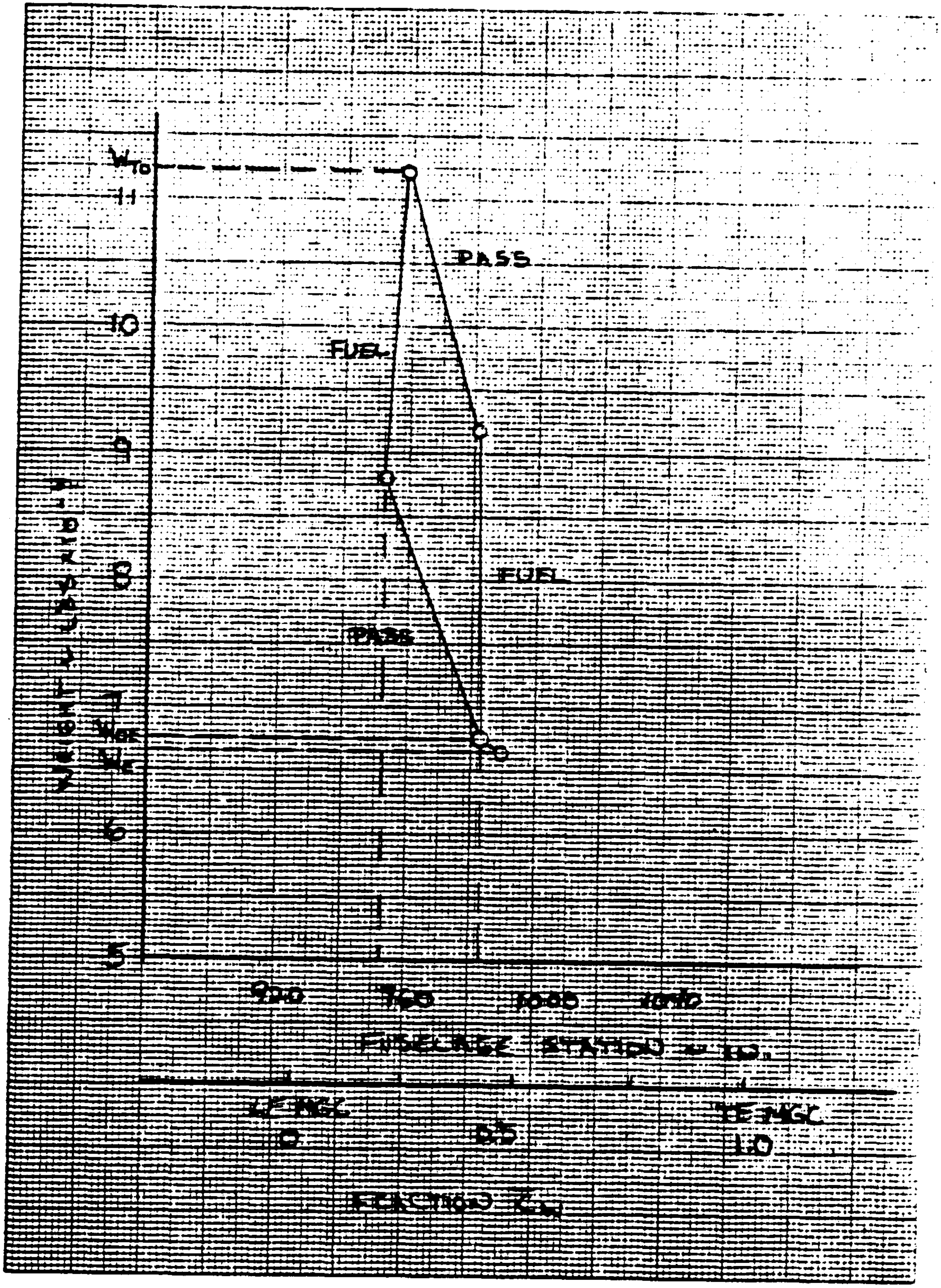

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FIGURE 3.7.4 CENTER OF GRAVITY EXCURSION DIAGRAM OF THE 100 PASSENGER MODEL

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Ref. 6 were used as supplements for these calculations. Table 3.7.5 contains geometric quantities and stability derivatives necessary to size the empennage for inherent stability. Design calculations are located in Appendix L, section L.7.

3.7.9.1 LONGITUDINAL STABILITY

From methods in Chapter 11 of Ref. 2, the horizontal tail was resized to incorporate a desired static margin of 5%. Appendix L, Figure L.2 presents the longitudinal x-plot for the 100 passenger airplane. From this plot, it is seen that a tail area of 155 ft^2 is required.

3.7.9.2 LATERAL - DIRECTIONAL STABILITY

From the method of Chapter 11 in Ref. 2, the vertical tail area required to hold engine-out flight was found to be critical. The engines were set at a five degree cant to lessen the thrust moment arm about the cg. The directional x-plot is given in Appendix L, Figure L.3. From this plot, the c_{n_β} at the required vertical tail area of 303 ft^2 was determined.

3.7.10 CLASS I DRAG POLARS

The Class I drag polars were calculated from the procedure of Chapter 12 in Ref. 2. The wetted areas of the airplane components were calculated as presented in Table 3.7.6 and Appendix L, section L.8. From the total airplane wetted area and assuming a skin friction coefficient of 0.0025, C_{D_o} for the airplane was calculated.

Table 3.7.7 contains the take-off, cruise, and landing drag polars computed during the initial performance sizing. These drag polars are compared to the drag polars computed from wetted area considerations. These Class I drag polars more accurately represent the airplane. Changes to C_{D_o} for take-off and landing conditions are given in Appendix L, section L.8.

The clean zero-lift drag coefficient at low speed was determined as:

$$C_{D_o} = 0.0115$$

The drag polars for take-off, landing, and cruise were then calculated as shown in Appendix L.

TABLE 3.7.5--STABILITY AND CONTROL RESULTS FOR THE 100 PASSENGER COMMUTER.

$S = 1604\ ft^2$; $\bar{c} = 11.6\ ft$; $b = 139\ ft.$

$\Delta \bar{x}_{ac_B} = -0.10$

$\bar{x}_{ac_{WB}} = 0.15$

$\bar{x}_{ac_A} = 0.506$ F.S. : 998

$\bar{x}_{ac_H} = 5.60$

$C_{L_{\alpha_W}} = 4.72\ rad^{-1}$

$C_{L_{\alpha_H}} = 3.67\ rad^{-1}$

$C_{L_{\alpha_V}} = 1.66\ rad^{-1}$

$C_{n_\beta} = 0.0655\ rad^{-1}$

$d\varepsilon/d\alpha = 0.162$

$\bar{x}_{cg_{aft}} = 0.454$ F.S. : 988

$x_V = 57.9\ ft.$

Assuming a $C_{L_{cr}} = 0.3$, the final drag polars yield:

$$(L/D)_{cr} = 20.4$$

During initial take-off weight sizing, $(L/D)_{cr}$ was assumed to be 16.

The sensitivities to take-off weight given in Table 3.7.3 show that:

$$\partial W_{TO}/\partial(L/D) = -5{,}067 \text{ lbs}$$

For the baseline configuration, this translates into:

$$(L/D)_{cr} = 20.4 - 16 = 4.4$$

$$W_{TO} = \Delta(L/D)_{cr}\ \partial W_{TO}/\partial(L/D) = -\ 22{,}295 \text{ lbs.}$$

Since the take-off weight is 112,288 lbs, the increase in lift-to-drag ratio causes a 20% decrease in take-off weight. According to Ref. 2, this percentage change in take-off weight indicates that the airplane needs to be resized with the initial weight sizing methods of Ref. 1.

TABLE 3.7.6--WETTED AREA BREAKDOWN.

	Component	S_{wet} (ft^2)
1.	Wing	3058
2.	Horizontal tail	320
3.	Vertical tail	626
4.	Fuselage	2937
5.	Engines	248
6.	Engine pylons	278
	Total	7467

$c_f = .0025$

$f = 18.5\ ft^2$

TABLE 3.7.7--DRAG POLAR COMPARISON.

Preliminary Results	Drag Polar	$(L/D)_{max}$
1. Clean	$0.0196 + 0.0312C_L^2$	20.2
2. Take-off, gear up	$0.0346 + 0.0332C_L^2$	14.7
3. Take-off, gear down	$0.0546 + 0.0332C_L^2$	11.7
4. Landing, gear up	$0.0946 + 0.0332C_L^2$	8.9
5. Landing, gear down	$0.1146 + 0.0332C_L^2$	8.1

$(L/D)_{cruise\ at\ C_L = 0.3} = 13.4$

Class I Results		
1. Clean	$0.0119 + 0.0312C_L^2$	25.9
2. Take-off, gear up	$0.0269 + 0.0332C_L^2$	16.7
3. Take-off, gear down	$0.0469 + 0.0332C_L^2$	12.7
4. Landing, gear up	$0.0869 + 0.0332C_L^2$	9.3
5. Landing, gear down	$0.1069 + 0.0332C_L^2$	8.4

$(L/D)_{cruise\ at\ C_L = 0.3} = 20.4$

3.7.11 CONCLUSIONS AND RECOMMENDATIONS

The following conclusions resulted from the preliminary design work on the NASA-100:

1. W_{TO} = 112,288 lbs; W_E = 66,042 lbs; W_{OE} = 67,422 lbs.

2. Powerplant: Two 13,500 lb turboprops, aft-mounted.

3. Commonality achieved:
   - a. fuselage cross-section.
   - b. cockpit layout.
   - c. landing gear.
   - d. natural laminar flow airfoils.

4. Achieved inherent longitudinal and directional stability.

5. The take-off weight will decrease by 20% due to high (L/D) characteristics, but it may increase due to the structural weight of high aspect ratio wings.

6. Wing-folding may need to be employed in order to meet existing gate requirements.

The following recommendations resulted from the preliminary design work:

1. The feasibility of folding the wings needs to be analyzed.

2. The 100 passenger airplane will need to be resized according to the methods of Ref. 1.

3. The feasibility of achieving a common wing torque box needs further study, but will be difficult to achieve.

4. This configuration should be replaced with the 100 passenger twin-body model. More commonality appears possible with the twin-body configuration. The twin-body model also has the advantage of a lighter take-off weight.

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4.0 COMPARISON OF COMMUTER FAMILY TO EXISTING AIRPLANES

The purpose of this chapter is to compare data from the commuter family with existing regional turbo-propeller driven airplanes. The larger members of the commuter family will be compared with smaller jet transports. Take-off weights, center of gravity excursion range, wetted areas, wing loadings and cabin and baggage volumes of the airplanes will be compared. These comparisons will attempt to prove the validity of the class I designs.

4.1 COMPARISON OF TAKE-OFF WEIGHTS

Figure 4.1 shows that the commuter family take-off weights compared with existing airplanes. The commuter family was sized assuming a 5% empty weight savings due to the use of advanced structural materials. Aramid Aluminum will be utilized to achieve this empty weight savings. Appendix E contains data for this composite material.

4.2 CENTER OF GRAVITY EXCURSION

Table 4.1 contains the excursion range of the center of gravity for the commuter family. These data are compared with common excursion ranges for regional turbo-propeller and jet transport airplanes taken from Reference 2.

From Table 4.1 it can be seen that all the class I designs have C.G. excursion ranges comparable with contempory airplanes. The large range of C.G. travel for the twin-body 75 passenger airplane is due to commonality constraints with the 36 passenger design.

4.3 COMPARISON OF AIRPLANE WETTED AREAS

Wetted areas of the commuter family are compared to regional turbo-propeller and jet transports wetted areas. Figure 4.2a compares the wetted areas of the smaller passenger capacity airplanes. Figure 4.2b compares the larger capacity airplanes. It can be seen that these airplanes compare favorably with existing regional turbo-propeller and jet transport airplanes.

4.4 COMPARISON OF AIRPLANE WING LOADINGS

Wing loadings of the commuter family are compared to existing commuters and jet transports. Table 4.2 lists wing loadings of some existing airplanes. Table 4.3 lists wing loadings for the commuter family. The comparison shows that the commuter family wing loadings are higher than typical commuters but less than jet transports.

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TABLE 4.1 CENTER OF GRAVITY EXCURSION RANGE COMPARISON

AIRPLANE MODEL	RANGE OF C.G. TRAVEL		COMMON EXCURSION RANGES	
25 passenger	21"	$.28\bar{c}$	12"-20"	$.14 - .27\ \bar{c}$
36 passenger	20"	$.25\bar{c}$	12"-20"	$.14 - .27\ \bar{c}$
50 passenger	15"	$.17\bar{c}$	12"-20"	$.14 - .27\ \bar{c}$
75 passenger	21"	$.17\bar{c}$	26"-91"	$.12 - .32\ \bar{c}$
100 passenger	30"	$.21\bar{c}$	26"-91"	$.12 - .32\ \bar{c}$
75 twin-body	31"	$.34\bar{c}$	26"-91"	$.12 - .32\ \bar{c}$
100 twin-body	16"	$.16\bar{c}$	26"-91"	$.12 - .32\ \bar{c}$

TABLE 4.2 WING LOADINGS OF EXISTING AIRPLANES

Airplane	$(W/S)_{TO}$ psf
CASA C-212-200	38.1
Shorts 330	50.5
Beech 1900	50.3
Fokker F27-200	59.7
DHC-6-300	29.8
DHC-7	66.5
DHC-8	52.1
EMB-120	51.7
BAe 31	53.9
METRO III	46.9
Fokker F-28	85.9
BAe 146-200	107.6

TABLE 4.3 WING LOADINGS FOR THE COMMUTER FAMILY

Airplane Model	$(W/S)_{TO}$ psf
25 Passenger	50
36 Passenger	70
50 Passenger	70
75 Passenger	70
100 Passenger	70
75 Twin-Body	84
100 Twin-Body	87

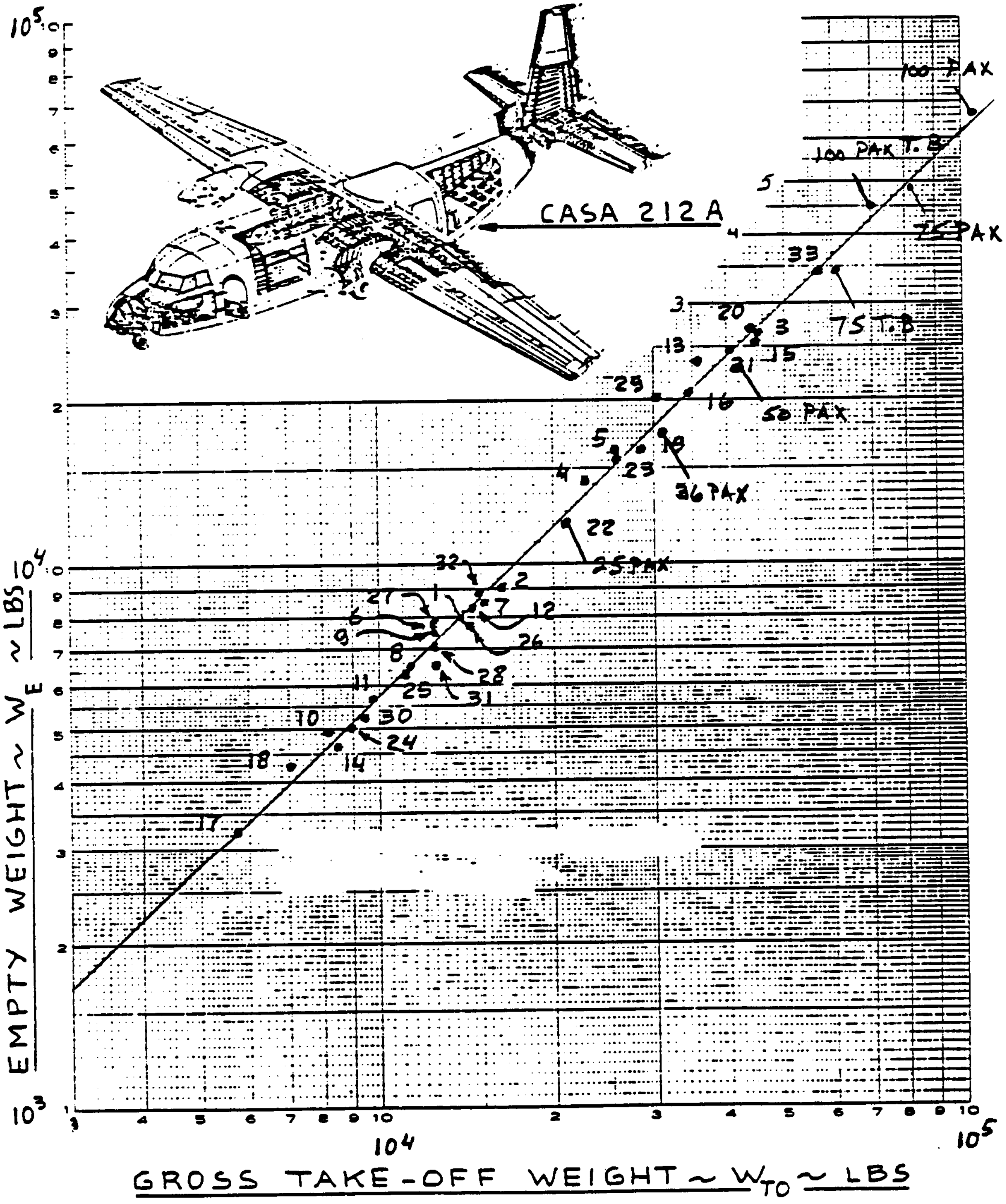
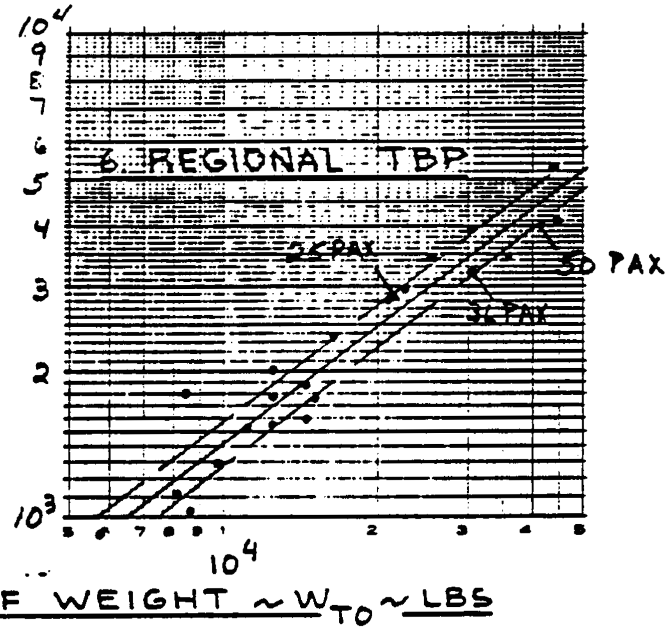


FIGURE 4.1 TAKE-OFF WEIGHT COMPARISON
Copied from Ref. 1.



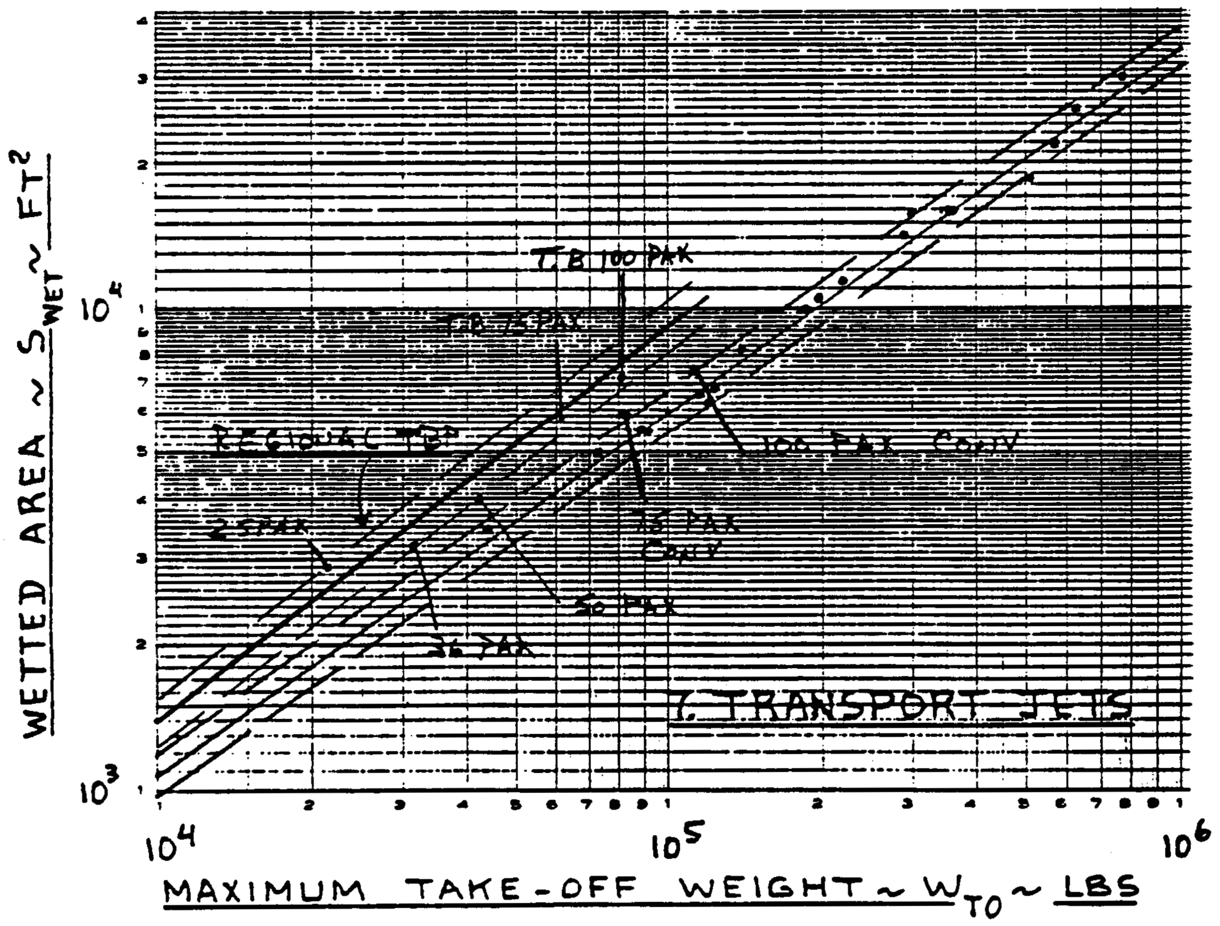


FIGURE 4.2 WETTED AREA COMPARISON
Copied from Ref. 1.

4.5 COMPARISON OF CABIN VOLUME WING EXISTING AIRPLANES

Passenger and baggage volume are compared with existing airplanes in Table 4.4. Data for Table 4.4 is compiled from Reference 8, Appendix B.

TABLE 4.4 COMPARISON OF CABIN AND BAGGAGE VOLUMES

Airplane Type	Number of Passengers	Overhead Baggage Volume (cuft)	Overhead Volume per Seat (cuft)
NASA			
50	50	56	1.1
36	36	41	1.1
25	25	29	1.2
British Aerospace			
BAe Super 748	46	41	0.85
BAe ATP	48	100	1.6
BAe 146-100	64	56	0.68
de Havilland			
DASH 7	50	59	1.2
DASH 8	37	32	0.86
Fokker			
F-27	52	40	0.77
50	50	79	1.6
F-28	65	107	1.6
Shorts			
330	30	40	1.3
360	36	52	1.4
ATR Consortium			
ATR 42-200	46	53	1.2
Embraer			
EMB-120	30	32	1.1

5.0 Commonality Analysis of the Commuter Family

Now that the Class I designs for the commuter family have been presented, the extent of commonality that was implemented needs to be discussed. Table 5.1 shows the status of the commonality objectives listed in Chapter 2.

The twin-body concept is extremely conducive to commonality implementation with the smaller commuters. This allows for more commonality throughout the passenger range.

The following items are common to all members of the commuter family:

1. Common fuselage cross section.
2. Common flight deck layout.
3. Common cockpit instrumentation.
4. Common landing gear tire sizes.

These features were implemented with a mimimum of configuration design problems.

To also achieve:

5. Common wing carry-thru structure.
6. Common landing gear retraction schemes,

the twin-body configurations were introduced. This allowed the above objectives to be integrated into the commuter family. The wing areas of the 75 and 100 passenger conventional configurations were too large to implement a common torque box carry-through structure. See Table 2.3. Also, the lateral gear spacing was too large to accommodate similar gear struts with the smaller members of the family. The 100 passenger conventional model has 4 tires per bogey on the main gear, while the twin-body 100 passenger only needed 2 wheels per bogey. See Table 2.4.

From reasons discussed in Appendix B, two different shp turbo-prop engines will be used to span the passenger models presented in Chapter 3. Table 5.1 shows what engines are integrated into the airplanes of the family.

From the Class I drag polar analysis conducted in Chapter 4, it was determined that to achieve the desired $(L/D)_{cr}$ values, the 12 aspect ratio wing will be needed. Therefore, the weight penalty of the wing design is necessary.

Empennage and tailcone commonality is desired. Design work necessary to complete a proposal for these items has not been completed yet. Handling qualities results and Class II weight and balance results will be required to submit a commonality proposal for the empennage and tailcone arrangement.

Systems commonality will require further study. For the flight control system design, the open loop handling qualities will be examined and common closed loop

Table 5.1--Status of Commonality in the Commuter Family.

Airplane Type	25 Pax	36 Pax	50 Pax	75 Pax	100 Pax	75 Pax Twin-Body	100 Pax Twin-Body
Structural Commonality:							
Fuselage Tailcone Arrangement			Further design work is required.				
Wing Torque Box	Yes	Yes	Yes	No	No	Yes	Yes
Fuselage Cross Section	Yes	Yes	Yes	Yes	Yes	Yes	Yes
Landing Gear	Yes	Yes	Yes	No	No	Yes	Yes
Systems Commonality:							
Cockpit Instrum.	Yes	Yes	Yes	Yes	Yes	Yes	Yes
Handling Qualities			Further design work is required.				
Fuel System							
De-Icing							
Pressurization							
Flight Controls							
Engine Commonality:							
2 Engines	Yes	Yes	Yes	Yes	Yes	Yes	Yes
6000 shp	Yes	Yes	Yes	No	No	No	No
13,500 shp	No	No	No	Yes	Yes	Yes	Yes

characteristics will be proposed. A separate surface stability augmentation system (SSSA) will be proposed. Also a fly by wire flight control system using electro-hydrostatic actuators will be researched.

The critical wing L.E. volume of the 25 passenger model will be implemented with a T.K.S. de-icing system. This system will then be able to fit into all the other airplanes in the family.

6. CONCLUSIONS AND RECOMMENDATIONS

6.1 CONCLUSIONS

1) A family of commuter airplanes have been desiged. These airplanes range from 25 to 100 passengers.

2) Take-off weights range from 21046 lbs to 112288 lbs.

3) The design of a commuter family of airplanes with commonality appears feasible if the twinbody concept is used.

4) Five designs have been selected to be taken through the class II design procedure:
    a) 25 passenger
    b) 36 passenger
    c) 75 twin-body
    d) 50 passenger
    e) 100 twin-body

5) The following commonality objectives have been integrated into the commuter family:

Common fuselage cross section
Common landing gear tire sizes
Common main and nose gear retraction schemes
Common wing torque boxes
Common powerplants (2)
Common cockpit instrumentation
Common NLF airfoil

6.2 RECOMMENDATIONS

1) Continue design work on the 25, 36, and 50 passenger models. The twinbody 75 and 100 passenger models should also be taken through some class II design methods.

2) Determine handling characteristics of the commuter family. This will allow for the design of a flight control system that will achieve handling commonality across the passenger range.

3) Propose a common empennage-tailcone arrangement.

4) Propose designs for common flight and operational systems.

7. REFERENCES

1) Roskam, J., Airplane Design: Part I, Preliminary Sizing of Airplanes. Roskam Aviation and Engineering Corporation, Route 4, Box 274, Ottawa, Kansas. 1985.

2) Roskam, J., Airplane Design: Part II, Preliminary Configuration Design and Integration of the Propulsion System. Roskam Aviation and Engineering Corporation, Route 4, Box 274, Ottawa, Kansas. 1985.

3) Roskam, J., Airplane Design: Part III, Cockpit and Fuselage Layouts. Roskam Aviation and Engineering Corporation, Route 4, Box 274, Ottawa, Kansas. 1985.

4) Roskam, J., Airplane Design: Part IV, Layout Design of Landing Gear and Systems. Roskam Aviation and Engineering Corporation, Route 4, Box 274, Ottawa, Kansas. 1985.

5) Roskam, J., Airplane Flight Dynamics and Automatic Flight Controls., Roskam Aviation and Engineering Corporation, Route 4, Box 274, Ottawa, Kansas. 1979.

6) Roskam, J., Methods for Estimating Stability and Control Derivatives of Conventional Subsonic Airplanes. Roskam Aviation and Engineering Corporation, Route 4, Box 274, Ottawa, Kansas. 1971.

7) Roskam, J., Airplane Design: Part V, Component Weight Estimation. Roskam Aviation and Engineering Corporation, Route 4, Box 274, Ottawa, Kansas. 1985.

8) Creighton, T.R., Marketing and Technology Survey of Commuter Airplanes. August 1, 1986.

APPENDIX A

COCKPIT AND FUSELAGE ARRANGEMENTS

TABLE OF CONTENTS

A.1 FUSELAGE CROSS SECTION

From Figure A.1 it is seen that many commuter airplanes in the 20 to 65 passenger range have 4-abreast seating. This range of passenger capacity spans over half of the required passenger capacity of the family. For this reason 4-abreast seating was selected.

Figure 2.1 shows the selected fuselage cross section to be used in all of the airplanes in the NASA commuter family. The overhead storage volume calculated in this section is compared with that of other commuter airplanes in tables A.1 and 4.4.

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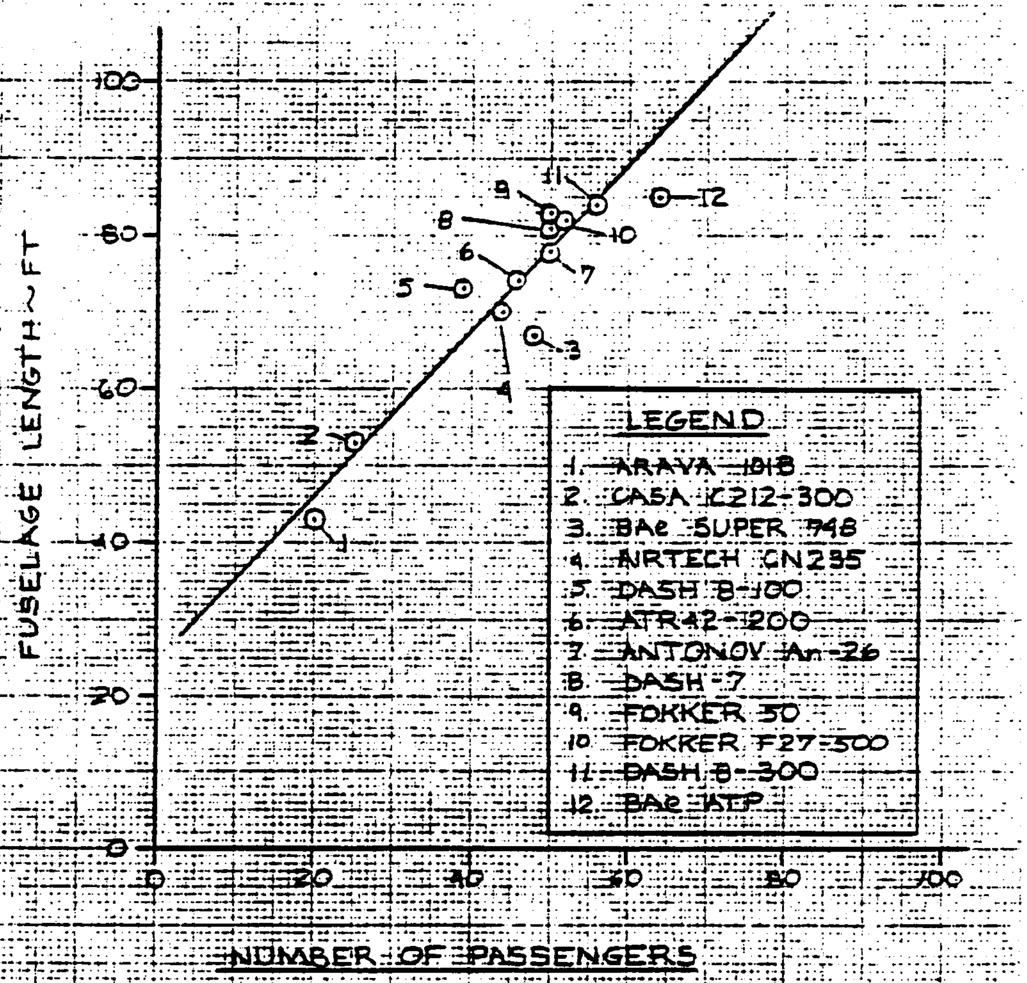
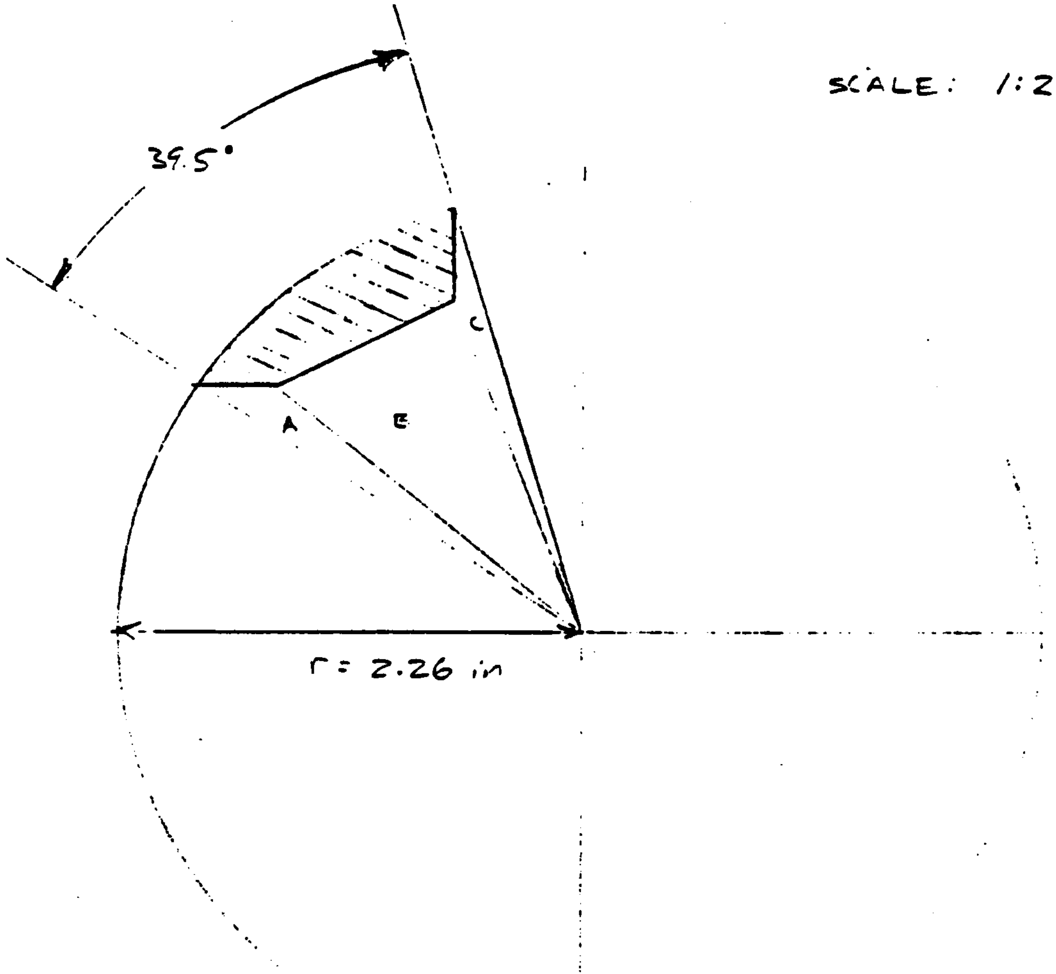


CALC			REVISED	DATE	FIGURE A.1: FUSELAGE TRENDS
CHECK					
APPD					
APPD					UNIVERSITY OF KANSAS

A.1.1 DETERMINATION OF OVERHEAD BAGGAGE VOLUME

OVERHEAD VOLUME

SCALE: 1:20 INCHES



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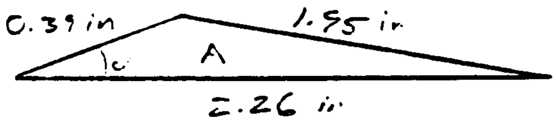
AREA OF SECTOR $= \frac{1}{2} r^2 \theta$ (θ in radians)

$= \frac{1}{2} (2.26)^2 (39.5^\circ)\left(\frac{\pi}{180}\right)$

$= 1.76 \text{ in}^2$

AREA OF TRIANGLE A:



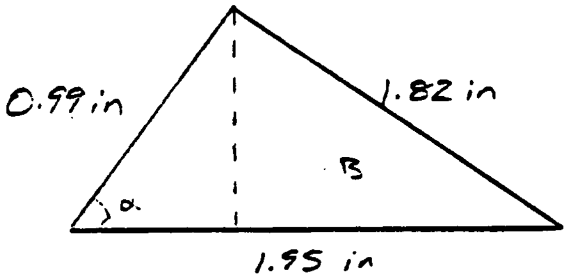
(NOT TO SCALE)

$\alpha = 34.2°$ $\therefore$ $h = 0.22$ in

$A = \frac{1}{2}bh = \frac{1}{2}(2.26)(0.22) = 0.25$ in^2

AREA A = 0.25 in^2

AREA OF TRIANGLE B:



(NOT TO SCALE)

$\alpha = 67.6°$ $\therefore$ $h = 0.92$ in

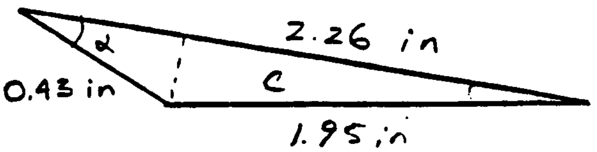
$A = \frac{1}{2}bh = \frac{1}{2}(1.95)(0.92) = 0.90$ in^2

AREA B = 0.90 in^2

AREA TRIANGLE C

$\alpha = 40°$ $\therefore$ $h = 0.28$ in

$A = \frac{1}{2}bh = \frac{1}{2}(2.26)(0.28) = 0.32$ in^2



(NOT TO SCALE)

AREA OF OVERHEAD STORAGE

$A = (1.76\ in^2) - (0.25\ in^2) - (0.90\ in^2) - (0.32\ in^2)$

$A = 0.29\ in^2$

$A = 116\ in^2 = 0.81\ ft^2$

50 PASSENGER OVERHEAD VOLUME

$$V = \frac{(0.81\ ft^2)(8.5\ in)(50\ in/in) + (0.81\ ft^2)(8\ in)(50\ in/in)}{(12\ in/ft)}$$

$$\boxed{VOLUME = 56\ FT^3}$$

36 PASSENGER OVERHEAD VOLUME

$$V = \frac{(0.81\ ft^2)(6.1\ in)(50\ in/in)(2\ rows)}{(12\ in/ft)}$$

$$\boxed{V = 41\ FT^3}$$

25 PASSENGER OVERHEAD VOLUME

$$V = \frac{(0.81\ ft^2)(4.3\ in)(50\ in/in)(2\ rows)}{(12\ in/ft)}$$

$$\boxed{V = 29\ FT^3}$$

FIGURE A.1 LISTS THE OVERHEAD VOLUME PER PASSENGER OF THE 25, 36 AND 50 PASSENGER COMMUTERS ALONG WITH THE VALUES FOR OTHER COMMUTER AIRPLANES FOR COMPARISON.

TABLE A.1 COMPARISON OF CABIN AND BAGGAGE VOLUMES

Airplane Type	Number of Passengers	Overhead Baggage Volume (cuft)	Overhead Volume per Seat (cuft)
NASA			
50	50	56	1.1
36	36	41	1.1
25	25	29	1.2
British Aerospace			
BAe Super 748	46	41	0.85
BAe ATP	48	100	1.6
BAe 146-100	64	56	0.68
de Havilland			
DASH 7	50	59	1.2
DASH 8	37	32	0.86
Fokker			
F-27	52	40	0.77
50	50	79	1.6
F-28	65	107	1.6
Shorts			
330	30	40	1.3
360	36	52	1.4
ATR Consortium			
ATR 42-200	46	53	1.2
Embraer			
EMB-120	30	32	1.1

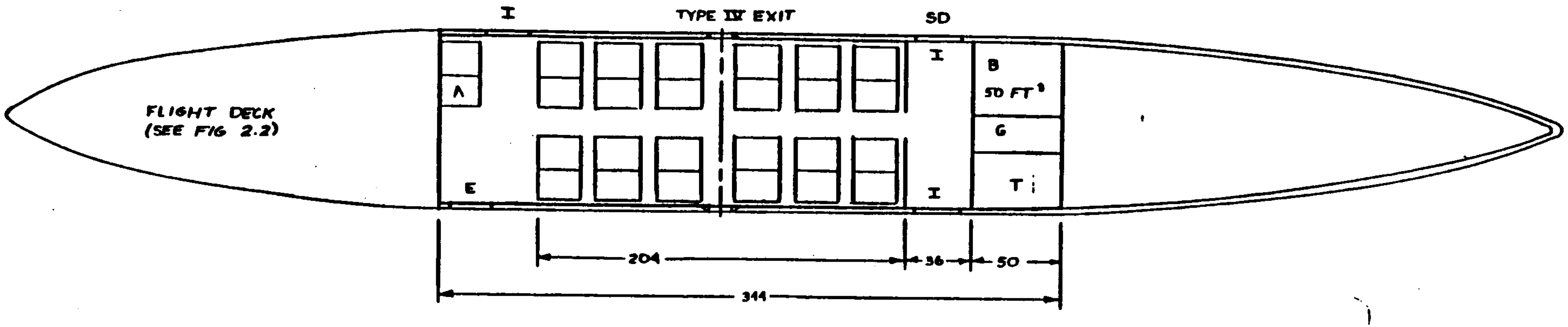
A.3 CABIN LAYOUTS

The cabin layouts presented in this section were 'laid out' using the methods presented in References (2) and (3). The seat pitch chosen was 32 inches which is consistent with those of other commuter airplanes as shown in Reference (8).

Figure A.2 presents the cabin layout for the 25-passenger commuter.

Figure A.3 presents the cabin layout for the 36-passenger commuter along with an alternate cockpit layout having 3 passenger seats to be used as the second cockpit on a twin body 75-passenger commuter.

Figure A.4 presents the cabin layout for the 50-passenger commuter.



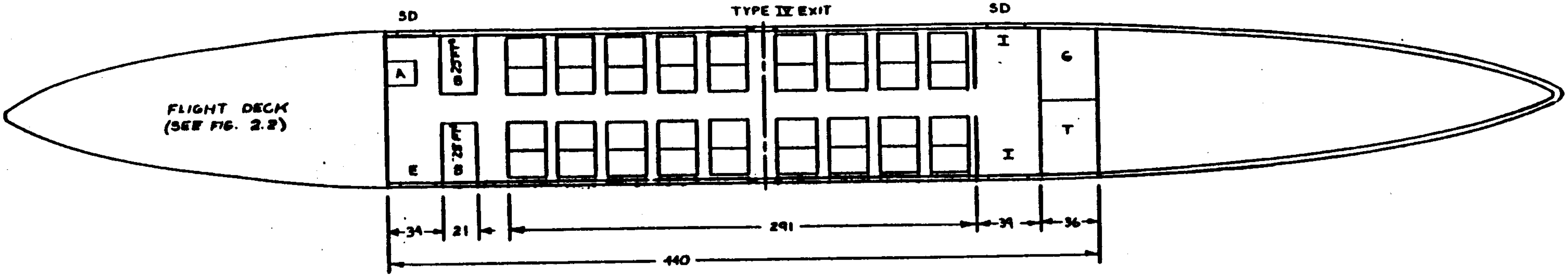
A: ATTENDANT SEAT
B: BAGGAGE COMPARTMENT
E: ENTRANCE DOOR

G: GALLEY
I: EMERGENCY DOOR
SD: SERVICE DOOR

T: TOILET

FIGURE A.2: 25 PASSENGER CABIN LAYOUT

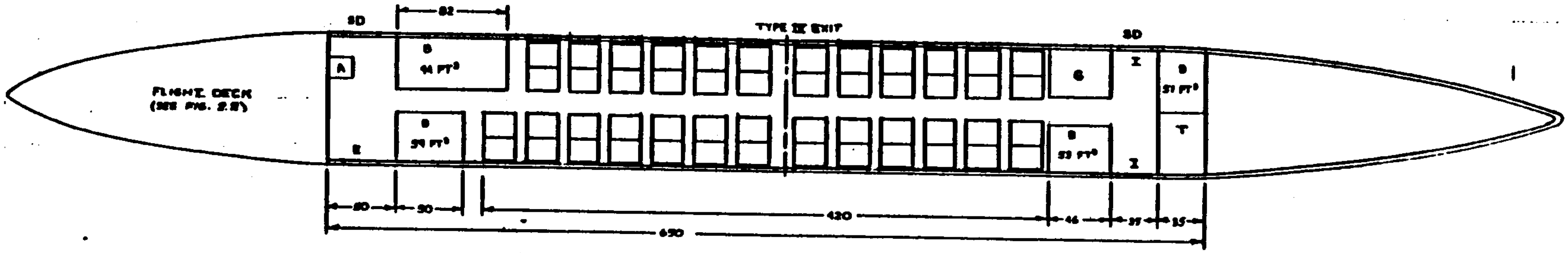
SCALE: 1:50 INCHES



A: ATTENDANT SEAT
B: BAGGAGE COMPAR
E: ENTRANCE DOOR
G: GALLEY
I: EMERGENCY DOOR
SD: SERVICE DOOR

FIGURE A.3: 36 PASSENGER CABIN LAYOUT

SCALE: 1:50 INCHES



A: ATTENDANT SEAT
B: BAGGAGE COMPARTMENT
E: ENTRANCE DOOR

G: GALLEY
I: EMERGENCY DOOR
SD: SERVICE DOOR

T: TOILET

FIGURE A.11 50 PASSENGER CABIN LAYOUT

SCALE: 1:50 INCHES

APPENDIX B

Advanced Counter-rotation Propfan Engine Data

Statement of Purpose:

The purpose of this appendix is to provide the engine data and configuration used throughout this study.

Table of Contents

1. Engine Data Source

The engine data used in this report are taken from ADVANCED PROPFAN ENGINE TECHNOLOGY (APET) AND SINGLE AND COUNTER-ROTATION GEARBOX/PITCH CHANGE MECHANISM. NASA CR-168115, by Allison Gas Turbine Division, General Motors Corporation.

The study engine falls under the designation PD436-11. The technology in this propulsion system is verifiable in the late 1980's and is appropriate for production in the mid 1990's.

Two engines for this study have been scaled from the APET report: a 6000 shp engine and a 13,500 shp engine. The baseline engine is shown in Figure B.1.

2. Engine Selection Criteria

The initial criteria proposed for selecting a propulsion system for the commuter family was as follows:

* 2 powerplants per airplane
* aft-mounted pusher configurations
* one common engine core used throughout

Due to the wide range of power levels required between the 25 and 100 passenger airplanes (4210 - 13400 shp), it was decided to use two different engine cores:

6000 shp engine core:	for the 25, 35, and 50 passenger configurations
13,500 shp engine core:	for the 75 and 100 passenger twin body configurations

Obviously, the 25 passenger design will be overpowered by 30 percent, but the engine can be "flat-rated" to meet the airplane's maximum needs. This means the 25, 36, and 75 passenger designs will carry an extra weight penalty.

3. 6,000 SHP Engine Data

Dimensions:

Overall length	108.5 inches
Maximum height	35.4 inches
Maximum width	26.2 inches
Maximum engine diameter	24.9 inches
Reduction gearbox diameter	36.4 inches

Weight:

Engine weight	879	lbs
Reduction gearbox and interconnecting structure	308	lbs
Propeller weight	1690	lbs
Nacelle weight	964	lbs

Performance:

Sea level, standard day at maximum power

Power = 6204 shp
sfc = 0.368 lbs/hp/hr

4. 13,500 SHP Engine Data

Dimensions:

Overall length	150.1	inches
Maximum height	48.9	inches
Maximum width	36.2	inches
Maximum engine diameter	37.3	inches
Reduction gearbox diameter	54.6	inches

Weight:

Engine weight	1,995	lbs
Reduction gearbox and interconnecting structure	1,040	lbs
Propeller weight	1,690	lbs
Nacelle weight	1,360	lbs

Performance:

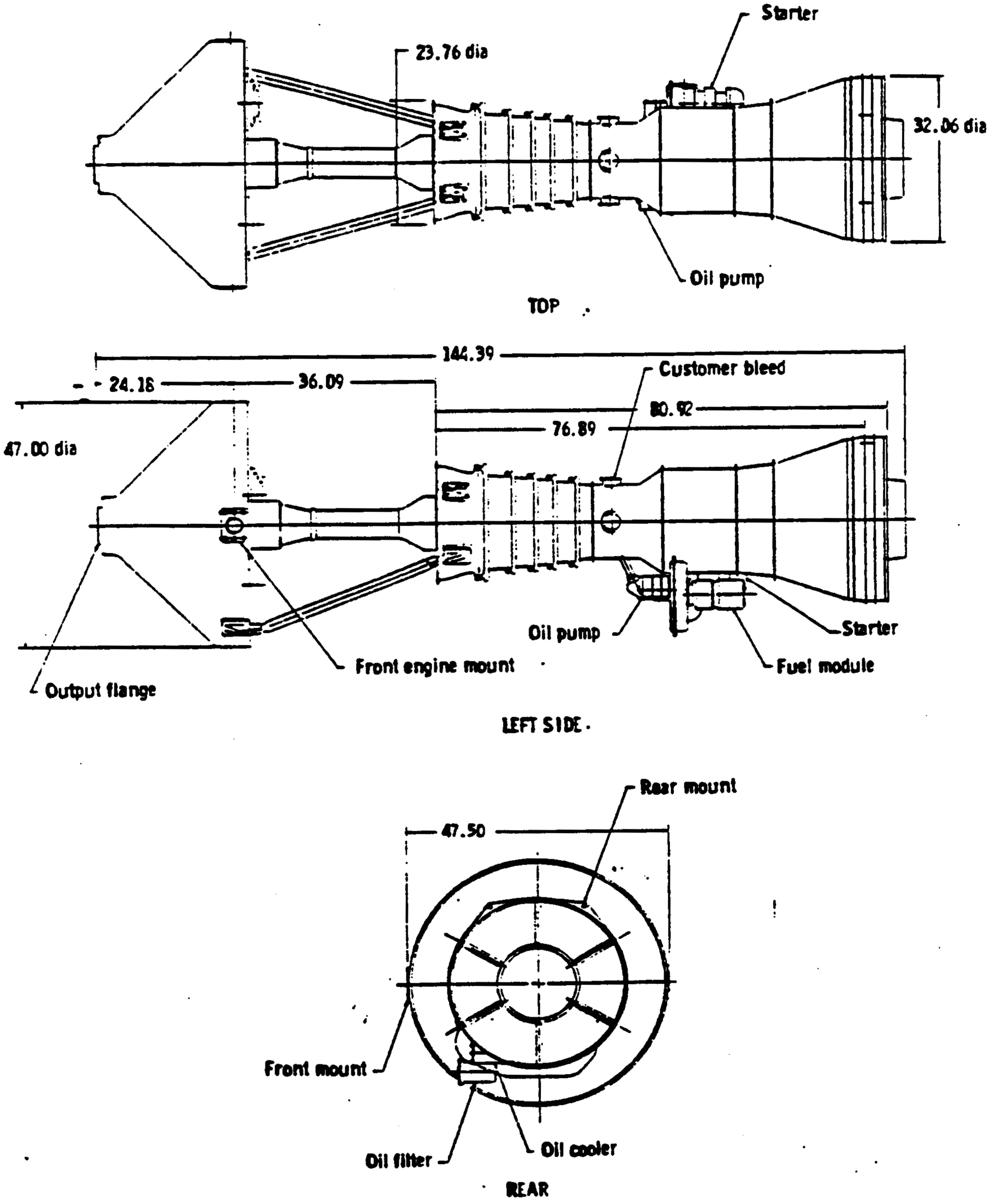
Sea level, standard day at maximum power

Power = 13,457 shp
sfc = 0.357 lbs/hp/hr

5. Installation Characteristics

The following dimensions are related to Figure B.2. The installation data is for a counter-rotation pusher propfan (6x6 blades) for $M_{cruise} = 0.70$.

$L_s = 0.55D$ where, D - Blade diameter
$L_{cg} = 0.09D$ BL - Blade length
$d = 0.25D$
$F_{bf} = 1.5BL$



Note: All dimensions are in inches

TE83-2243

Figure B.1 PD436-11 Powerplant

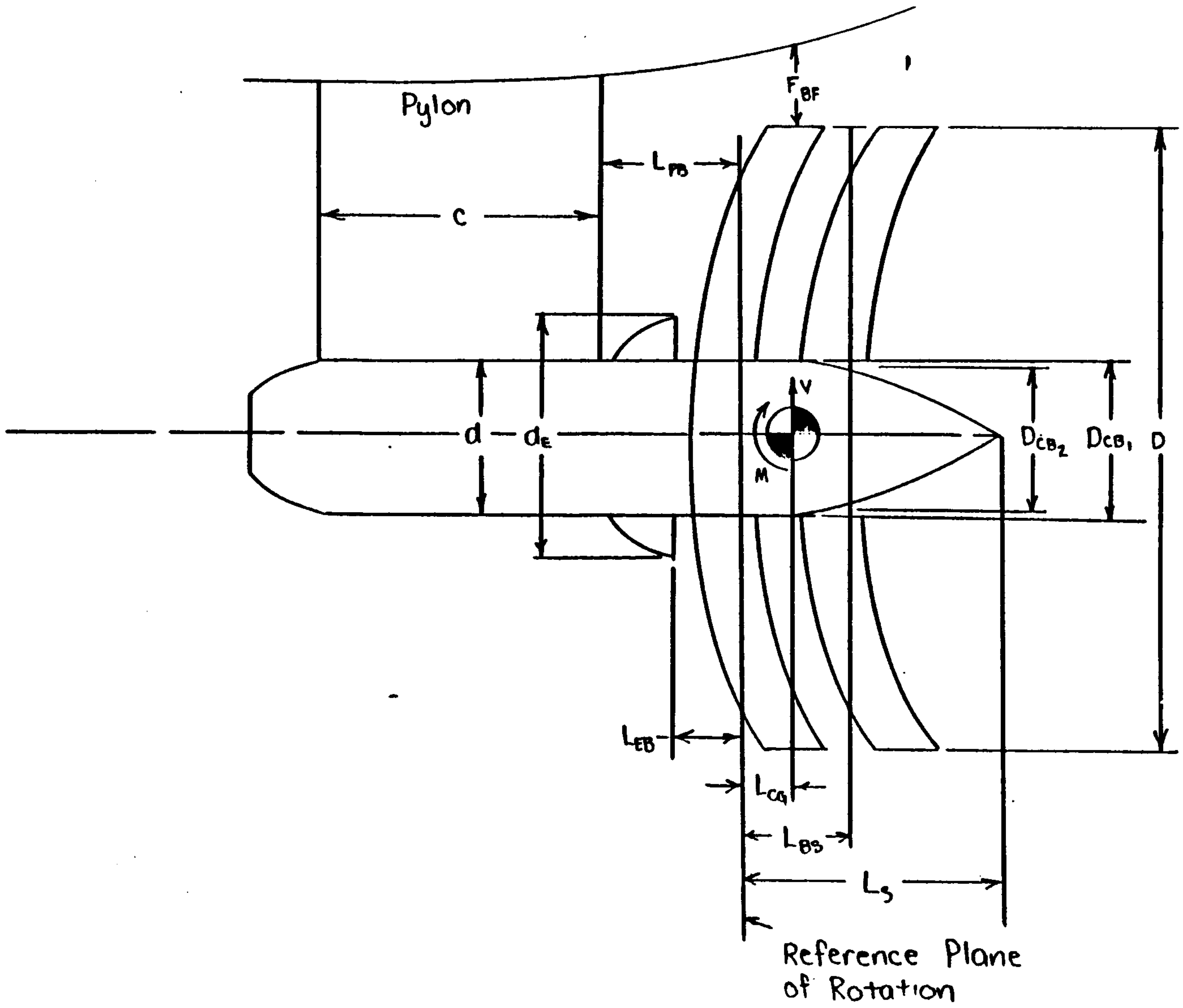


Figure B2. CR Pusher Propfan Installation Parameters.

APPENDIX C

AIRFOIL DATA

C.1 SUMMARY

This appendix details the procedure, and decisions made in determining realistic NLF airfoil section data. The design conditions for the airfoil are:

1) Drag Divergence Mach Number of .75
2) Design Lift Coefficient of .40

The airfoil section described herein is a paper airfoil. It is modeled after the HSNLF(1)-0213 airfoil designed by J. Viken at NASA Langley. To obtain actual data, extensive computer analysis and wind tunnel tests would be needed, which are beyond the scope of this project.

The assumed airfoil characteristics are:

$t/c = .117$ (unswept)

$t/c = .129$ (swept 20°)

$C_{l_{max}} = 1.6$

$C_{d_{min}} = .0035$ (low speed)

$C_{d_{min}} = .0075$ (cruise $M = .70$)

$C_{l_\alpha} = .105 \text{ deg}^{-1}$

$C_{m_{ac}} = -.10$

$M_{dd} = .75$

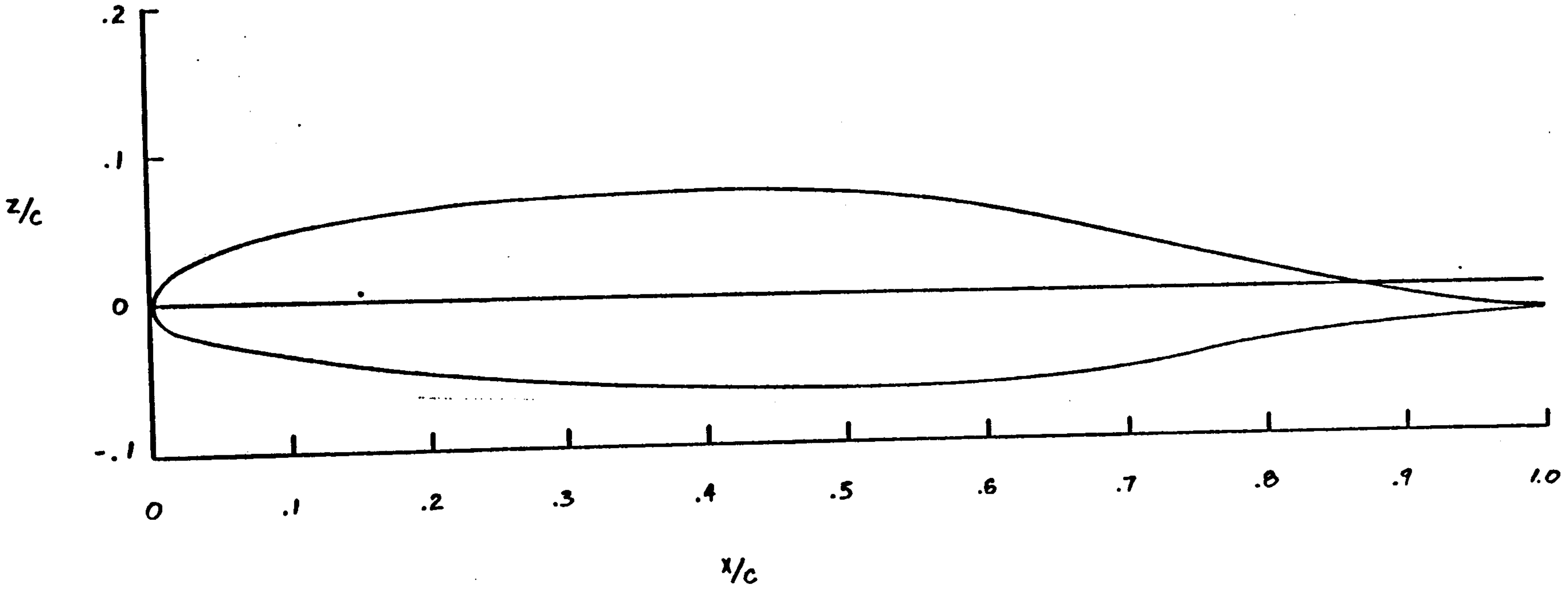


FIGURE C.1 - PROPOSED NLF AIRFOIL CROSS SECTION

Appendix D

Class I Landing Gear Retraction Scheme

D. Class I Landing Gear Retraction Scheme

This section presents the retraction kinematics for the landing gear of the family of commuter transports.

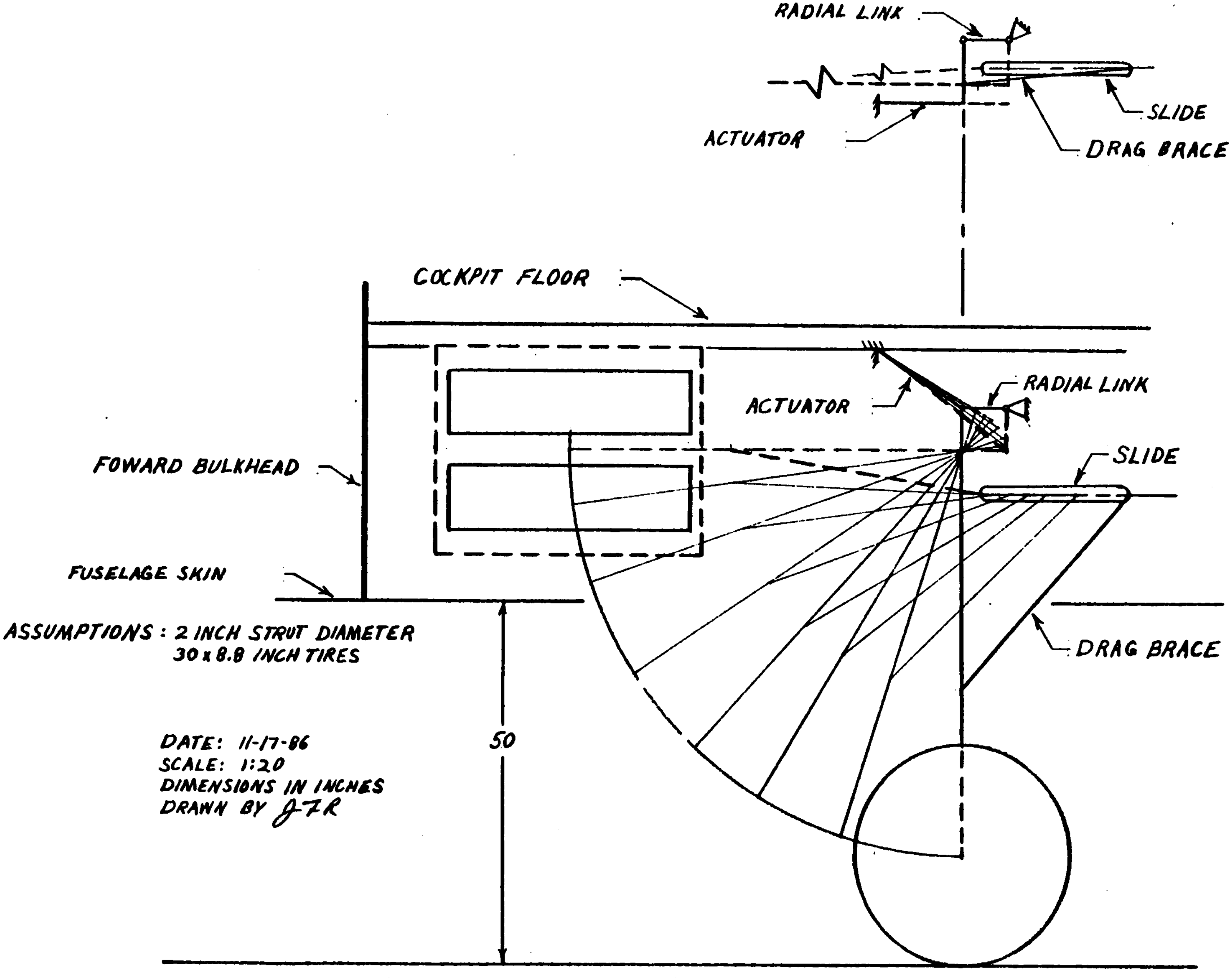
From Reference 2 a preliminary tire choice was made with the following dimensions:

Do = 30 inches   W = 8.8 inches

To achieve complete stowage of the nose gear a retraction scheme which incorportated the tires turning 90 degrees relative to the main strut was required. Figure D.1 shows the retraction kinematics for the nose gear.

The main gear could not be stowed within the fuselage. Figure D.2 shows the retraction kinematics for the main gear and the modification made to house the main gear.

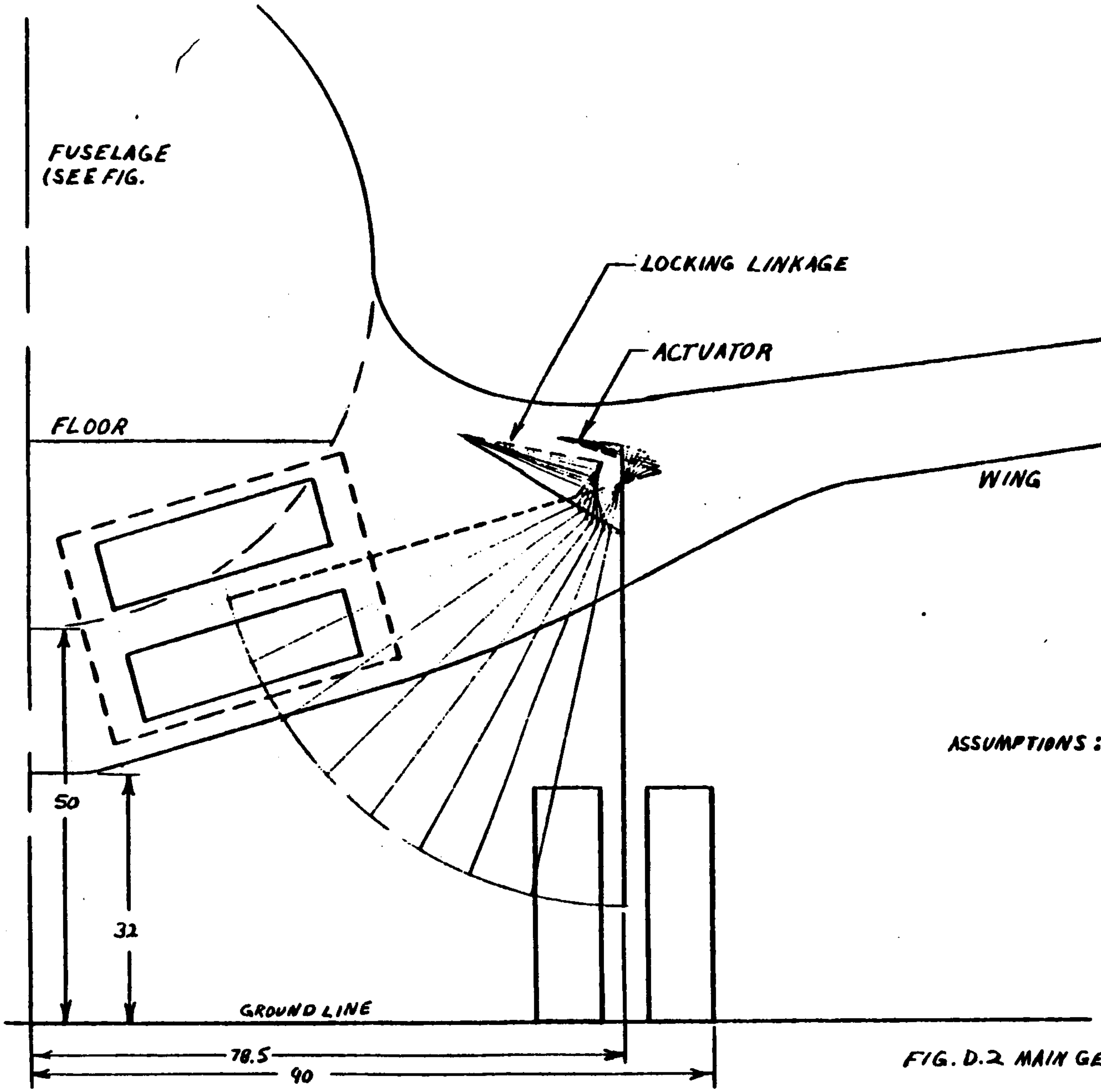
The Class II landing gear analysis may result in some changes to the landing gear as proposed here. These changes are believed to be, increase the number of tires on the main landing gear from two to four or use two different tires, one for the nose gear and one type for the main gear.



ASSUMPTIONS : 2 INCH STRUT DIAMETER
30 x 8.8 INCH TIRES

DATE: 11-17-86
SCALE: 1:20
DIMENSIONS IN INCHES
DRAWN BY JFR

FIG. D.1 NOSE GEAR RETRACTION KINEMATICS



ASSUMPTIONS : 4 INCH STRUT DIAMETER
30 x 8.8 INCH TIRES
4.5 INCH RADIAL CLEARANCE
2 INCH TIRE-WALL CLEARANCE

DATE : 11-16-86
SCALE: 1:20
DIMENSIONS IN INCHES
DRAWN BY JTR

FIG. D.2 MAIN GEAR RETRACTION KINEMATICS

APPENDIX E

ARAMID ALUMINUM DATA SUMMARY

Table of Contents

September 4, 1986

Preliminary Overview of Feasibility of using ARALL as a Primary Component of Aircraft Structures

ARALL - Aramid Aluminum Laminate, based upon an August 1983 report.

E.1 PROPERTIES:

	2024T3	7075T6	ARALL*
.2% Yield Stress (KSI)	52	70	77
Ultimate Tensile Stress (KSI)	68	81	114
Proportional Limit Comp. (KSI)	39	70	47
Youngs Modulus (KSI)	10440	10440	9135
Failure Strain %	17	11	3.5
Specific Weight	2.8	2.8	2.45
Density lb/ft^3	174.8	174.8	152.95

*ARALL 7075-T6 sheets with intermediate modulus fibers and pre-strained.

E.2 STRENGTHS:

High static strength particularly in tensile yield stress.

High fatigue resistance, in fact it is almost fatigue insensitive, with a life cycle of a factore of ten(10) times more testing cycles.

Better corrosion resistance, including the bondline when pretreated.

Delamination under heavy loads and corrosive environment is no problem.

Quality control by C-scan and Fokker bond tester easily detected delamination and voids.

E.3 MACHINABILITY:

Easily cut, drilled, sawn and milled by normal workshop procedures.

Countersinking is possible with conventional rivets. Briles rivets are ideal for thin skin installation.

Adhesive bonding with pretreatment and high temperature curing is allowable.

This material can also be bolted.

Plastic sheet bending is possible, including fabrication of stiffeners and limited double curvature bending.

E.4 AREAS OF CONCERN:

Prestressing of fibers, a technique to obtain better compressive properties, is "rather expensive".

Strength decreases with moisture absorption. Stiffness is not significantly affected.

Notched fracture toughness is comparable or worse than Al alloy. (Intermediate modulus fibers had best properties when notched)

Low fracture toughness when through the thickness damage(cut fibers) occurred. Although it had far superior fracture toughness with the fibers intact. This is offset by whether such accidental damage will ever occur.

Avoid peel forces higher than 0.146 psf.

E.5 MOST LIKELY STRUCTURAL COMPONENT USES:

Where panelloading is above 6.27 psf, probably in lower skin of wing

cylindrical part of pressure cabin

Lower Wing: Changes from fatigue critical to mainly critical in compression(negative gust case).

Fuselage has two critical areas:

Bottom: Fatigue critical in tangential; compression critical in axial.

Crown: Fatigue critical.

Overall, where used yielded about 30 percent decrease in structural weight.

APPENDIX F

CLASS I WEIGHT FRACTIONS FOR THE COMMUTER FAMILY

F.1 STATEMENT OF PURPOSE

The purpose of this appendix is to present the class I weight fractions for the airplane family components. These weight fractions were compiled from weight data in Reference 7. Table F.1 displays the airplanes used to compile the database and the weight fractions for the commuter family.

TABLE F.1 CLASS I WEIGHT FRACTIONS

Component	Fokker F-27-200	Fokker F-27-500	DeHavilland DHC7-102	 DHC6-300	Commuter Family
Fuselage	.099	.114	.106	.136	.114
Wing	.104	.100	.111	--	.105
Empennage	.024	.024	.030	.024	.025
Powerplant	--	--	.107	.100	.103
Landing Gear	.042	.041	.039	--	.041
Fixed Eqpt.	--	.144	.169	.145	.153

APPENDIX G

WING TORQUE BOX COMMONALITY

G.1 STATEMENT OF PURPOSE

The primary objective of this Appendix is to determine the location of the front spar and the rear spar such that the chord lengths of the wing torque boxes of the 25, 36, and 50 passenger airplanes are equal in length. Table G.1 lists the wing geometries.

TABLE G.1 COMMUTER FAMILY WING GEOMETRIES

	25 pax	36 pax	50 pax
Wing Area ft^2	421	449	591
Aspect Ratio	12	12	12
Wing Span ft	71.1	73.4	84.2
Root Chord ft	8.46	8.74	10.0
Taper Ratio	0.40	0.40	0.40

Of these different wing configurations the length of the torque box was limited by the wing root chord length of the 25 passenger commuter. The results are listed in Table G.2. See Figure 2.4 for the wing overlays with the common torque box structure shown.

TABLE G.2 WING SPAR LOCATION

Passenger	Front Spar		Rear Spar	
Model	Root	Tip	Root	Tip
25	.080	.080	.850	.850
36	.075	.084	.790	.785
50	.110	.130	.750	.615

APPENDIX H

ENGINEERING CALCULATIONS FOR THE 25 PASSENGER COMMUTER

TABLE OF CONTENTS

H.1 Introduction

The purpose of this Appendix is to present the preliminary sizing and Class I design calculations for the 25 passenger commuter. Methods used were taken from References 1 and 2. References 5 and 6 were used for stability and control design calculations.

Section H.2 contains preliminary weight sizing calculations. These results are from XEWTO6, a computer program available at Kansas University.

Section H.3 contains preliminary performance results from XPRFRM, a computer interactive program available at the University of Kansas.

Section H.4 contains Class I flap sizing calculations.

Section H.5 contains Class I empennage sizing ($\bar{V}$-method)

Section H.6 contains landing gear design criteria.

Section H.7 contains stability and control calculations.

Section H.8 contains the wetted area calculations and the Class I Drag Polars.

H.2 Initial Weight Sizing

Using XE WTOG, a weight sizing program which follows the method in Ch 2 of REF 1, the following weights and take-off weight sensitivities for the 25 passenger airplane were determined.

See Table I.1.

The design assumptions used in the weight sizing were:

$(L/D)_{CR} = 16$

$c_p = .4$ lb/Hp/Hr

$\eta_p = .85$

$V_{CR} = 442$ knts.

TABLE H.1 Initial Weight Sizing Results with Sensitivities

```
WEIGHT ESTIMATION FOR A REGIONAL,
PROPELLER DRIVEN AIRCRAFT
PASSENGER WEIGHT IS      5125.
CARGO WEIGHT IS             0.
CREW WEIGHT IS         410.
PHASE W/W  CJ CR CP    NP      L/D      ALTCR       RC    MCR OR V    E OR R    PLDROP
  1   0.990    0.00    0.00    0.00      0.         0.0     0.00        0.00       0.
  2   0.995    0.00    0.00    0.00      0.         0.0     0.00        0.00       0.
  3   0.995    0.00    0.00    0.00      0.         0.0     0.00        0.00       0.
  4   0.985    0.00    0.00    0.00      0.         0.0     0.00        0.00       0.
  5   0.906    0.40    0.85   16.00      0.         0.0     0.00     1100.00       0.
  7   0.985    0.00    0.00    0.00      0.         0.0     0.00        0.00       0.
  8   0.995    0.00    0.00    0.00      0.         0.0     0.00        0.00       0.
REGRESSION COEFFICIENTS ARE A=0.3989 AND B=0.9647
THE MISSION FUEL FRACTION WITHOUT RESERVES IS:0.857
THE GROSS TAKE OFF WEIGHT IS      21500. POUNDS.
THE EMPTY WEIGHT IS        11953. POUNDS.
THE WEIGHT OF FUEL IS        3845. POUNDS.


**EMPTY WEIGHT REDUCTION DUE TO COMPOSITES: 5.0 PER CENT
** SENSITIVITY ANALYSIS BEGINS HERE **
GROWTH FACTOR DUE TO PAYLOAD WEIGHT IS..........            4.2
THE TAKE-OFF WEIGHT TO EMPTY WEIGHT SENSITIVITY IS          1.7


CHOICE NUMBER..         5
                    CRUISE
SFC (LB/LB/HR)                ****
SFC (LB/HP/HR)                0.40
PROP EFFICIENCY               0.85
L/D                           16.0
VELOCITY (KNOTS)               0.0
RANGE (NT. MILES)           1100.0
ENDURANCE (HRS)            *******
THE SENSITIVITY OF GROSS TAKE-OFF WEIGHT TO THE FOLLOWING PARAMETERS
IS NOW GIVEN AS THE PARTIAL DERIVATIVE OF THE GROSS TAKE-OFF WEIGHT
TO THE INDICATED PARAMETER.
DWTO/DCJ (LB/LB/LB/HR)             0.0
DWTO/DCP (LB/LB/HP/HR)         20556.2
DWTO/DNP (POUNDS)              -9861.8
DWTO/D(L/D) (POUNDS)            -523.9
DWTO/DV (LB/KNOT)                  0.0
DWTO/DR (LB/NT MILE)               7.6
DWTO/DE (LB/HR)                    0.0
```

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H 3 Initial Performance Sizing

The results from XPRFRM, a performance sizing program, are presented in this section. The methods used are presented in Ch 3 of Ref 2.

The results are presented in Tables H.3 through H.6.

TABLE H.2

```
**************************************************
***************   TAKE-OFF SIZING   ***************
**************************************************

          "FAR 25 CERTIFICATION CATEGORY"
                      REGIONAL TURBO-PROP
                    PROPELLER DRIVEN
------------------  INPUT DATA  ------------------

ALTITUDE                                 0.0(FEET)
FAR 23 TAKE-OFF DISTANCE <STO>           0.0 (FEET)
MINIMUM WING LOADING                   20.00(LB/FT**2)
MAXIMUM WING LOADING                  100.00(LB/FT**2)
MINIMUM TAKE-OFF LIFT COEFFICIENT 1.40
MAXIMUM TAKE-OFF LIFT COEFFICIENT 2.60
-------------------  OUTPUT DATA  ----------------

         TABLE OF POWER LOADINGS

 W/S             CLMAX-TO

0.00      1.40    1.80    2.20    2.60

 20.0     18.8    24.2    29.6    35.0
 40.0      9.4    12.1    14.8    17.5
 60.0      6.3     8.1     9.9    11.7
 80.0      4.7     6.1     7.4     8.8
100.0      3.8     4.8     5.9     7.0
```

TABLE H.3 Landing Field Length Sizing

```
                    REGIONAL TURBO-PROP
                FAR 25 CERTIFICATION CATEGORY
GROSS TAKE-OFF WEIGHT  (WTO)      21046.0(LBS)
LANDING TO TAKE-OFF WEIGHT RATIO      1.000
ALTITUDE                              0.0(FEET)
DENSITY                           .0023769(SLUG/FT**3)
LANDING APPROACH SPEED  (VA)        108.0(KTS)
LANDING FIELD LENGTH   (SFL)       3500.0(FEET)

      (W/S)TO= 23.40CLMAX(LAND)

       MAXIMUM TAKE-OFF WING LOADINGS
   TO MEET LANDING DISTANCE REQUIREMENT
  ====================================

    CLMAX         MAXIMUM WING LOADING

    (LAND)            (TAKE-OFF)
                      (LB/FT**2)
   --------      -------------------

     1.40              32.77
     1.80              42.13
     2.20              51.49
     2.60              60.85
```

TABLE H.4

```
***************************************************
**************  DRAG POLAR EQUATIONS  ************
***************************************************


            ******  INPUT DATA  ******

     MAXIMUM TAKE-OFF WEIGHT (CLEAN)     21046.0  (LBS)
     WING AREA                             420.00  (FT**2)
     ASPECT RATIO                           12.00
     SKIN FRICTION COEFFICIENT            0.00250
     AIRPLANE WETTED AREA                2910.(FT**2)

  DRAG INCREMENT DUE TO TAKE-OFF FLAPS  .0200
  DRAG INCREMENT DUE TO LANDING FLAPS   .0600
  DRAG INCREMENT DUE TO LANDING GEAR    .0150

  OSWALDS EFFICIENCY FACTOR (CLEAN)     .850
  OSWALDS EFFICIENCY FACTOR (TAKE-OFF)  .800
  OSWALDS EFFICIENCY FACTOR (LANDING)   .800

            ******  CALCULATED DATA  ******

  THE COMPLETE SET OF DRAG POLARS IS:

     1.  LOW-SPEED  (CLEAN):
     CD =  .0208 + .0312CL**2        L/Dmax =19.63

     2.  TAKE-OFF   (LANDING GEAR UP):
     CD =  .0408 + .0332CL**2        L/Dmax =13.60

     3.  TAKE-OFF   (LANDING GEAR DOWN):
     CD =  .0558 + .0332CL**2        L/Dmax =11.63

     4.  LANDING    (LANDING GEAR UP):
     CD =  .0808 + .0332CL**2        L/Dmax = 9.66


     5.  LANDING    (LANDING GEAR DOWN):
     CD =  .0958 + .0332CL**2        L/Dmax = 8.87

****************************************************
```

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TABLE H.5 a)

```
FAR 25.111 (CEI)  "INITIAL CLIMB SEGMENT"
=========================================

FAR 25.111 CLIMB GRADIENT (INITIAL SEGMENT)   1.2000
               TABLE OF POWER LOADINGS REQUIRED
               --------------------------------

WING LOADING =  20.00      40.00      60.00      80.00     100.00
  (LB/FT**2)

ASPECT
RATIO

10.00           72.72      51.42      41.99      36.36      32.52
11.00           75.43      53.34      43.55      37.72      33.74
12.00           77.85      55.05      44.95      38.93      34.82
13.00           80.03      56.59      46.20      40.01      35.79
14.00           81.99      57.97      47.34      40.99      36.67
```

TABLE H.5 b)

```
FAR 25.121 (CEI)  "SECOND SEGMENT CLIMB"
========================================

FAR 25.121 CLIMB GRADIENT (SECOND SEGMENT)   2.4000
               TABLE OF POWER LOADINGS REQUIRED
               --------------------------------

WING LOADING =  20.00      40.00      60.00      80.00     100.00
  (LB/FT**2)

ASPECT
RATIO

 10.00          64.77      45.80      37.40      32.39      28.57
 11.00          66.92      47.32      38.63      33.46      29.53
 12.00          68.81      48.66      39.73      34.41      30.77
 13.00          70.50      49.85      40.71      35.25      31.53
 14.00          72.02      50.93      41.58      36.01      32.21
```

TABLE H.5c)

```
FAR 25.121 (CEI)  "TRANSITION SEGMENT CLIMB"
============================================

FAR 25.121 CLIMB GRADIENT (TRANSITION)        0.1000
               TABLE OF POWER LOADINGS REQUIRED
               --------------------------------

WING LOADING =  20.00      40.00      60.00      80.00     100.00
  (LB/FT**2)

ASPECT
RATIO

 10.00          58.27      41.20      33.64      29.14      26.06
 11.00          61.62      43.57      35.56      30.81      27.56
 12.00          64.72      45.76      37.37      32.36      28.94
 13.00          67.60      47.80      39.03      33.80      30.23
 14.00          70.28      49.69      40.57      35.14      31.43
```

TABLE H.5 d)

FAR 25.121 (OEI) "EN-ROUTE CLIMB SEGMENT"

FAR 25.121 CLIMB GRADIENT (EN-ROUTE) 1.2000

TABLE OF POWER LOADINGS REQUIRED

WING LOADING = (LB/FT**2) ASPECT RATIO	20.00	40.00	60.00	80.00	100.00
10.00	91.83	64.94	53.02	45.92	41.07
11.00	95.76	67.71	55.29	47.88	42.83
12.00	99.30	70.22	57.33	49.65	44.41
13.00	102.51	72.49	59.19	51.26	45.84
14.00	105.43	74.55	60.87	52.72	47.15

TABLE H.5 e)

FAR 25.119 (AEO) "LANDING CLIMB SEGMENT"

FAR 25.119 CLIMB GRADIENT (LANDING) 3.2000

TABLE OF POWER LOADINGS REQUIRED

WING LOADING = (LB/FT**2) ASPECT RATIO	20.00	40.00	60.00	80.00	100.00
10.00	26.14	18.48	15.09	13.07	11.69
11.00	26.95	19.05	15.56	13.47	12.05
12.00	27.66	19.56	15.97	13.83	12.37
13.00	28.29	20.00	16.33	14.14	12.65
14.00	28.85	20.40	16.66	14.43	12.90

TABLE H.5 f)

FAR 25.121 (OEI) "GO-AROUND OR BALKED LANDING"

FAR 25.121 CLIMB GRADIENT (GO-AROUND) 2.1000

TABLE OF POWER LOADINGS REQUIRED

WING LOADING = (LB/FT**2) ASPECT RATIO	20.00	40.00	60.00	80.00	100.00
10.00	41.93	29.65	24.21	20.97	18.75
11.00	42.74	30.22	24.67	21.37	19.11
12.00	43.43	30.71	25.08	21.72	19.42
13.00	44.04	31.14	25.42	22.02	19.69
14.00	44.57	31.52	25.73	22.28	19.93

TABLE H.6

```
                   POWER LOADINGS NECESSARY
          TO MEET THE CRUISE SPEED REQUIREMENTS
==================================================================

  (W/S)       (W/S)        (W/P)        (W/P)        (W/P)
 ACTUAL      TAKEOFF      ACTUAL       TAKEOFF      TAKEOFF
                                      IN FLIGHT      STATIC
  (psf)       (psf)       (lb/hp)      (lb/hp)      (lb/hp)
------------------------------------------------------------------

    20.00       20.00         2.51         2.51         0.75   .93
    40.00       40.00         5.02         5.02         1.51  1.86
    60.00       60.00         7.53         7.53         2.26  2.79
    80.00       80.00        10.03        10.03         3.01  3.71
   100.00      100.00        12.54        12.54         3.76  4.64
```

H.4 Flap Sizing

Using the method of Ref 2, Ch. 7, it was determined that the following flap geometry would supply the incremental lift necessary for take-off and landing (Table H.7)
The design calculations are included.

TABLE H.7 25 Passenger Flap Geometry

Trailing Edge Fowler Flaps.

$c_f/c = .15$

$S_{wf}/S = .9$

$b_f/b = .9$

$\delta_f = 25^\circ$

Class I Flap Sizing.

From Ch 7, Ref 2

$C_{L_{max}} = 1.4$

$C_{L_{max_{TO}}} = 1.4$

$C_{L_{max_L}} = 2.2$

Using Eqn 7.1

$C_{L_{max_w}} = 1.05\, C_{L_{max}} = 1.47$

Using Eqn 7.2 to correct for wing sweep:

$C_{L_{max_w}} = C_{L_{max_{w_{unswept}}}} \cos \Lambda_{c/4}$

$= 1.47 \cos 13^\circ = 1.43$

$C_{L_{max_w}} = 1.43$

Using Eqn. 7.3 for airfoil sectional $C_{l_{max}} = 1.5$

$K_\lambda = .95$ for $\lambda_w = .4$

$C_{L_{max_w}} = K_\lambda \, (C_{l_{max_r}} + C_{l_{max_t}})/2$

$C_{L_{max_w}} = 1.42$

The results of Eqns 7.2 and 7.3 are less than 1 percent different, therefore to be conservative:

$\boxed{C_{L_{max_w}} = 1.43}$

The required incremental $C_{L_{max}}$ to be generated by flaps:

Take-off $\Delta C_{L_{max_{TO}}} = 1.05\,(C_{L_{max_{TO}}} - C_{L_{max}})$

$\Delta C_{L_{max_{TO}}} = 0$

Landing $\Delta C_{L_{max_L}} = 1.05\,(C_{L_{max_L}} - C_{L_{max}})$

$\Delta C_{L_{max_L}} = .84$

Using Eqn 7.9,

$$\Delta C_{L_{max}} = \Delta C_{l_{max}} (S/S_{wf}) K_\Lambda$$

From Eqn 7.9, $K_\Lambda = .94$

$$\Delta C_{L_{max}} = .79 (S/S_{wF})$$

where $S/S_{wf} = .9$, which requires full span flaps over the wetted wing,

$$\boxed{\Delta C_{l_{max}} = .88}$$

Fowler flaps were chosen to generate this $\Delta C_{l_{max}}$.

For ΔC_l of the flap section, by Eqn 7.11

$$\Delta C_l = (1/K) \Delta C_{l_{max}}$$

$C_{l_\alpha} = 6.6$ rad^{-1} (NLF airfoil data)

$c_f/c = .25$

$C_{l_{\alpha_f}} = 6.6$ rad^{-1}

by Eqn 7.14

$$\Delta C_l = C_{l_{\alpha_f}} \alpha_{\delta_f} \delta_f$$

for $\delta_f = 25°$, $\alpha_{\delta_f} = .34$

from Figure 7.8

$$\Delta C_l = 6.6(.34)(25/57.3)$$

$$\boxed{\Delta C_l = .98}$$

Comparing this with the result of Eqn 7.11, it is seen that the choices for the flaps will generate the required ΔC_l.

H.5 $\bar{V}$-Bar Method for Empennage Sizing

Ch 8 Ref 2

A T-tail type Empennage, Conventional.

Table H.5 [illegible] of the [illegible].

[illegible] average value for $\bar{V}_H$, $\bar{V}_V$ from Tables 8.6a and 8.6b Resulted in:

$\bar{V}_H = 1.08$

$\bar{V}_V = .083$

Similarly for S_e/S_H, S_a/S, S_r/S_V averages were found,

$S_e/S_H = .36$

$S_a/S = .06$

$S_r/S_V = .34$

for the 25 pax

$S = 421\ ft^2$

$\bar{c} = 6.29\ ft$

$X_H = 31.8\ ft.$

$b = 71.1\ ft.$

$X_V = 27.1\ ft.$

from Eqn 8.3 of Ref 2

$S_H = \bar{V}_H\ S\ \bar{c} / X_h$

$S_H = 61\ ft^2$

From Eqn 8.4, Ref 2

$S_V = \bar{V}_V\ S\ b / X_V$

$S_V = 56\ ft^2$

For the control surfaces

$S_a = 29\ ft^2$

$S_r = 19\ ft^2$

$S_e = 22\ ft^2$

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Table H.8 Geometry for the Empennage

Horizontal Tail	Vertical Tail
$S_H = 61\ ft^2$	$S_V = 56\ ft^2$
$A_H = 3.5$	$A_V = 1.1$
$\lambda_H = .65$	$\lambda_V = .8$
$\Lambda_{LE} = 20°$	$\Lambda_{LE} = 45°$
$\bar{C}_H =$	$(t/c)_V = .11$
$b_H = 14.6\ ft.$	$\Gamma = 90°$
$C_r = 6.4\ ft.$	$i = 0°$
$C_t = 4.16\ ft.$	$b_V = 7.9\ ft.$
$(t/c) = .11$	$C_r = 8.0\ ft.$
$\Gamma = 0°$	$C_t = 6.4\ ft.$
$i = 0°$	$\bar{C}_V = 6.8\ ft.$

H.6 Class I Landing Gear Design

From Ch 9 of Ref 2, it was decided to choose a 30" dia tire by 9" wide. This tire can carry 20 000 lb.

From weight and balance calculations, longitudinal gear placement criterion were met. There is 15° between the ground contact point and the aft c.g. location.

Figure H.1 shows that lateral tip-over criteria is met for a 270" wheel base

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Lateral Tip-over Criteria Test

for the 25 pax

To satisfy requirements, angle ψ must be ≤ 55°

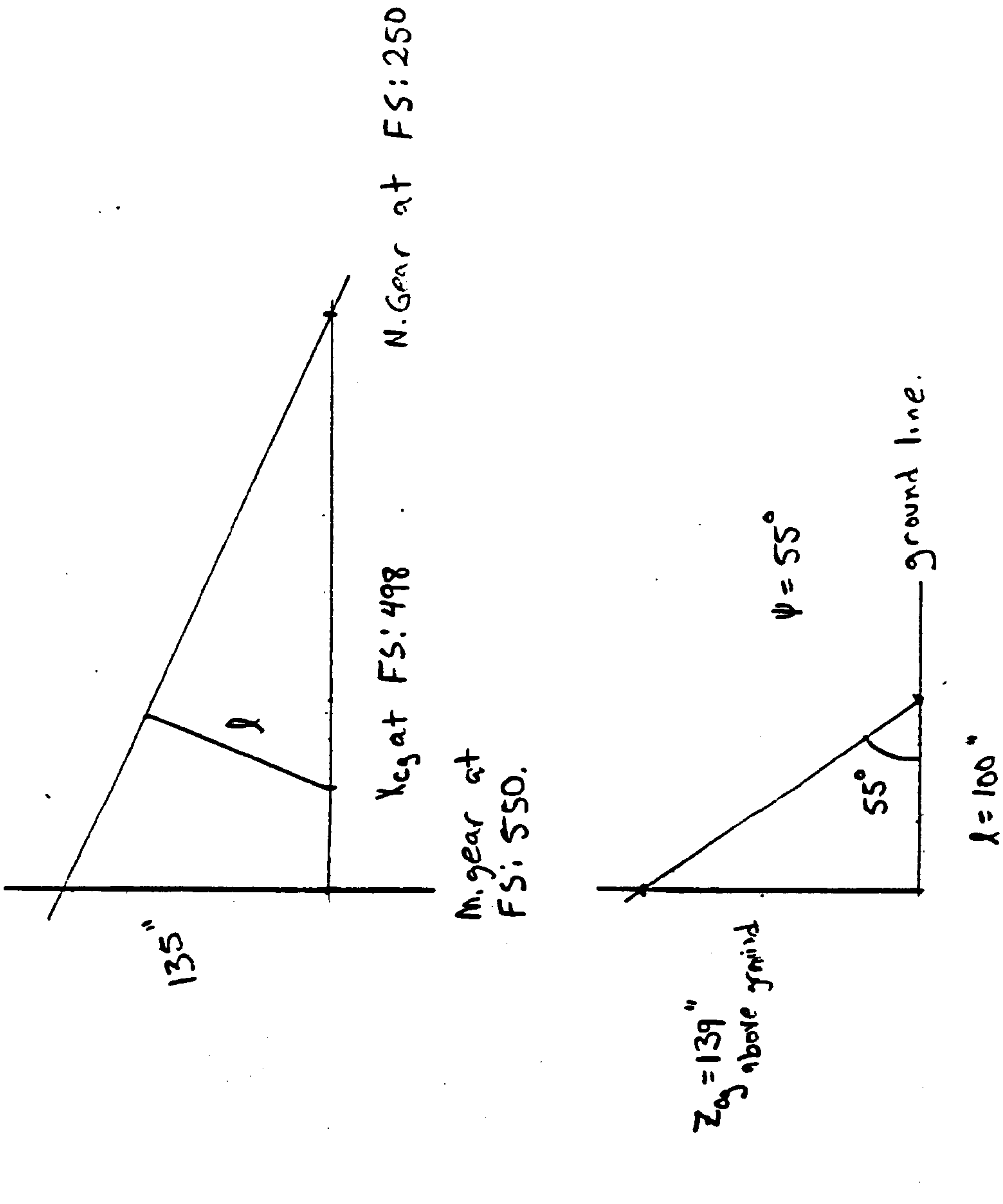

FIGURE H.1 Lateral Tip-Over Criteria

H.7 Stability And Control Calculations.

Calculation of required Stability derivatives are Presented in this Section.

25 Pax

$C_{L_{\alpha w}}$, Wing Lift Curve Slope.

$M = .7$ $\quad S = 421\ ft^2$

$\Lambda_{LE} = 15° = .2618$ rad

$A = 12$

From Ref 5, fig 3.12

For $A: 12$

$K = 1 + [(8.2 - 2.3\,\Lambda_{LE}) - (.22 - .153\,\Lambda_{LE})A]/100$

$K = 1.0544$

$\Lambda_{c/2} = 11.12°$ $\qquad \tan \Lambda_{c/2} = .1965$

$C_{l_\alpha} = 6.0\ rad^{-1}$ (NLF Airfoil data)

$\beta = \sqrt{1 - M^2} = .7141$

$k = C_{l_\alpha} / 2\pi/\beta = .6820$

$$C_{L_{\alpha w}} = \frac{2\pi A}{\left(2 + \sqrt{\frac{A^2 \beta^2}{k^2}\left(1 + \frac{\tan^2 \Lambda_{c/2}}{\beta^2}\right) + 4}\right) K}$$

$$\boxed{C_{L_{\alpha w}} = 4.71\ rad^{-1}}$$

$C_{L_{\alpha H}}$, Horizontal tail lift curve slope.

$\Lambda_{LE} = 20°$ $\quad \Lambda_{c/2} = 14.9°$ $\quad \tan \Lambda_{c/2} = .2699$

$S = 69\ ft^2$ $\quad A = 4$ $\quad K = 1.0673$

$\boxed{C_{L_{\alpha H}} = 3.41\ rad^{-1}}$ using Fig 3.12 of Ref 5.

Calculation of $d\epsilon/d\alpha$

Using Ref 5.

From figure 3.75

$M =$

$\boxed{d\epsilon/d\alpha = .220}$

$(1 - d\epsilon/d\alpha = .78$

MULTHROP'S INTEGRATION (125 PASSENGER)

METHOD: REF. 5 SECTION 3.4.6

$C_f = 82''$ $S = 420.9 \text{ FT}^2$

$l_H = 415''$ $\bar{c} = 6.28 \text{ FT}$

$\left.\frac{d\varepsilon}{d\alpha}\right._H = .220$

$C_{L_{\alpha_{WB}}} = C_{L_{\alpha_W}} = 4.71 \text{ RAD}^{-1}$

$$\overline{\Delta X}_{AC_B} = \frac{-\frac{dM}{d\alpha}}{\bar{q} S \bar{c} C_{L_{\alpha_W}}}$$ EQN 3.31 REF 5

$$-\frac{dM}{d\alpha} = \frac{\bar{q}}{36.5} \sum w_f^2(x_i) \left.\frac{d\bar{\varepsilon}}{d\alpha}\right|_i \Delta x_i$$ EQN 3.28a REF 5.

	x_i	Δx_i	w_f	w_f^2	$\frac{d\varepsilon}{d\alpha}$	$\frac{d\bar{\varepsilon}}{d\alpha}$	$\frac{dM}{d\alpha}\left(\frac{36.5}{\bar{q}}\right)$
1	332	62	65	4225	1.03	1.06	176.3
2	236	70	92	8464	1.05	1.08	370.3
3	165	79	96	9216	1.07	1.10	463.5
4	90	70	96	9216	1.12	1.15	429.3
5	34	55	96	9216	2.65	2.72	797.7
6	77	153	96	9216	—	.14	114.2
7	215	119	92	8464	—	.40	233.1
8	317	118	56	3136	—	.60	128.5
N1	125	120	30	900	—	.24	15.0
N2	125	120	30	900	—	.24	15.0

$\Sigma = 2743$

$$\frac{dM}{d\alpha} = \frac{\bar{q}}{36.5}(2743) = 75.2\,\bar{q}$$

$$\Delta \bar{X}_{ac_B} = \frac{-\frac{dM}{d\alpha}}{\bar{q}\,S\,\bar{c}\,C_{L_{\alpha_W}}} = \frac{-75.2\,\bar{q}}{\bar{q}\,(420.9)(6.28)(0.082)} = \boxed{-0.34}$$

$$\bar{X}_{ac_{WB}} = \bar{X}_{ac_W} + \Delta \bar{X}_{ac_B} = .25 - .34$$

$$\boxed{\bar{X}_{ac_{WB}} = -0.09}$$

Calculation of $\bar{X}_{ac_A}$

$\bar{X}_{ac_{WB}} = -.09$ $d\epsilon/d\alpha = .22$

$C_{L_{\alpha H}} = 3.41 \text{ rad}^{-1}$ $\bar{X}_{ac_H} = 6.30$

$C_{L_{\alpha WB}} = 4.71 \text{ rad}^{-1}$

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$$\bar{X}_{ac_A} = \frac{\bar{X}_{ac_{WB}} + \left\{C_{L_{\alpha H}}(1-d\epsilon/d\alpha)(S_H/S)\,\bar{X}_{ac_H}\right] / C_{L_{\alpha WB}}}{1 + \left\{C_{L_{\alpha H}}(1-d\epsilon/d\alpha)(S_H/S)\right] / C_{L_{\alpha WB}}}$$

This is Eqn 11.1 of Ref 2

$\bar{X}_{ac_A} = .45$

Writing $\bar{X}_{ac_A}$ in terms of S_H/S results in,

$$\bar{X}_{ac_A} = \frac{-.09 + .0085(S_H/S)}{1 + .0013(S_H/S)}$$

S_H	S_H/S	$\bar{X}_{ac_A}$
20	.048	.08
40	.095	.24
60	.143	.39
80	.190	.53
100	.238	.67

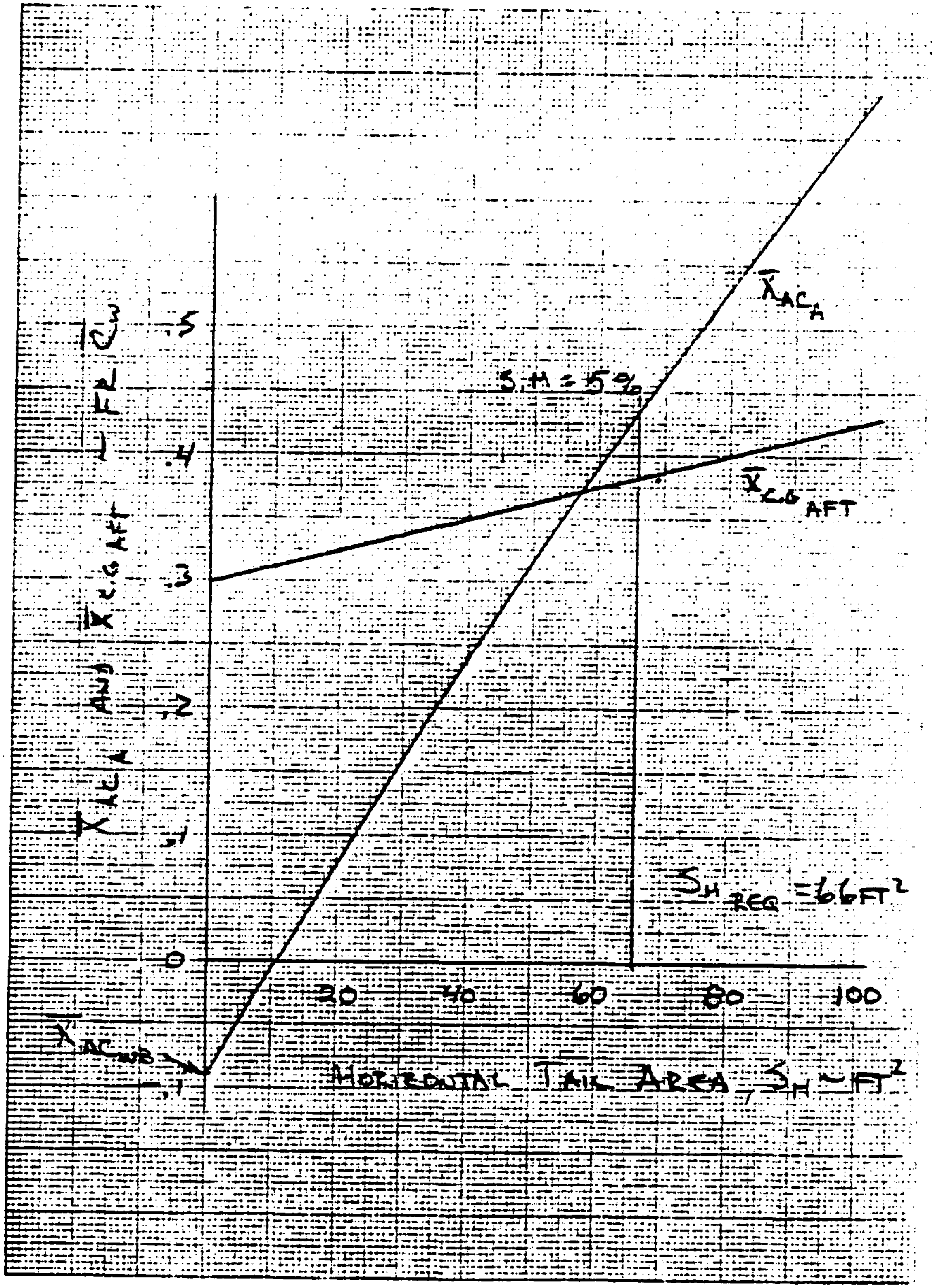
Plotted in Fig H.2

$\bar{X}_{cg}$ Shift due to changing tail area,

$W_H/S_H = 2.81 \text{ lb/ft}^2$

S_H	W_H	W_{Emp}	X_{cg}	$\bar{X}_{cg}$
50	140.6	520.2	422	.36
60	168.7	548.3	423	.37
70	196.8	576.4	423	.38

plotted in Fig H.2



CALC	10-28-86	TRC	REVISED	DATE	FIGURE H.2 LONGITUDINAL X-PLOT 25 PASSENGER COMMUTER	
CHECK						
APPD						
APPD						
					UNIVERSITY OF KANSAS	PAGE H-25

Calculation of $C_{L_{\alpha}V}$

S =

$C_{L_{\alpha}V}$ was calculated using Fig 3.12 of Ref 5.

The vertical tall geometry of Table H.5 was used as input data.

The result was,

$$C_{L_{\alpha}V} = 1.46 \text{ rad}^{-1}$$

$C_{n_{\beta_B}}$ Calculations FROM REF 6.

$$C_{n_{\beta_B}} = -57.3\, K_N K_{R_l} \frac{S_{B_S}}{S} \frac{l_B}{b} \left(\text{rad}^{-1}\right)$$

$l_B = 823 \text{ in} = 68.6 \text{ ft}$

$x_m = 423.2 \text{ in} = 35.3 \text{ ft}$

$w = h = 96.6 \text{ in} = 8.05 \text{ ft}$

$S_{B_S} = 62938.5 \text{ in}^2 = 437.1 \text{ ft}^2$

$\frac{l_B}{4} = 205.8 \text{ in} \implies h_1 = 8.05 \text{ ft}$

$\frac{3 l_B}{4} = 617.3 \text{ in} \implies h_2 = 5.83 \text{ ft}$

$\frac{x_m}{l_B} = 0.51$

$\frac{l_B^2}{S_{B_S}} = 10.8$

$\sqrt{\frac{h_1}{h_2}} = 1.2$

$\frac{h}{w} = 1$

$K_N = 0.0015$

$\mu = 3.106 \times 10^{-7}$ lb-sec/ft^2

$\rho = 0.8893 \times 10^{-3}$ slugs/ft^3

$V = 696.3$ ft/sec

$R_N = \frac{\rho V l_B}{\mu} = 136.8 \times 10^6$

$K_{R_l} = 2.005$

CALCULATION, FOR ENGINE-OUT FLIGHT, OF RUDDER DEFLECTION.

FROM REF 6.

$$C_{y_{\delta_R}} = -1.014 \frac{S_V}{S}$$

$$C_{n_{\delta_R}} = 1.014 \frac{S_V}{S} \frac{x_V}{b}$$

$$C_{n_{\delta_R}} = .428 \frac{S_V}{S}$$

WHERE:

$x_V = 30$ FT

$b = 71.1$ FT

$S = 420.9$ FT^2

$$\boxed{C_{n_{\delta_R}} = .001\ S_V}$$

FROM EQN 11.18 REF 2, Section 11.3

$$\delta_R = (N_D + N_{e_{CRIT}}) / \bar{q}_{mc}\, S b\, C_{n_{\delta_R}}$$

$y_{T_{ess}} = 7.5$ FT

$P_{TO_{req}} = 4210$ SHP - One Engine

$$T_{TO} = \frac{550\,(SHP)\,\eta_p}{V_{TO}}$$

$V_{TO} = 191$ FT/SEC

$\eta_p = .80$

$T_{TO} = 9698$ LBS

$N_{t_{crit}} = y_T\, T_{TO} = 72735$ FT LBS

$$C_{n_{\beta_B}} = -57.3\, K_N K_{RL} \frac{S_{BS}}{S} \frac{l_B}{b}$$

$$\boxed{C_{n_{\beta_B}} = -.171}$$

CALCULATION OF $C_{n_{\beta_V}}$

$$C_{n_\beta} = C_{L_\alpha V} \frac{S_V}{S} \frac{X_V}{b}$$ EQN 11.8 REF 2.

$C_{L_{\alpha_V}} = 1.46$ RAD^{-1}

$x_V = 30.0$ FT

$b = 71.1$ FT

$S = 420.9$ FT^2

$$\boxed{C_{n_{\beta_V}} = .0015\, S_V \text{ rad}^{-1}}$$

Calculation of C_{n_β}

$C_{n_\beta} = C_{n_{\beta_B}} + C_{n_{\beta_V}}$ Eqn 11 Ref 2

$$\boxed{C_{n_\beta} = -.171 + .0015\, S_V}$$

S_V	C_{n_β} (DEG^{-1})
0	-.0030
50	-.0017
100	-.0004
150	.0009

$S_{V_{REQ}} = 160$ FT^2 TO SATISFY LATERAL DIRECTIONAL REQUIREMENT FOR C_{n_β}

A DIRECTIONAL X-PLOT IS PRESENTED IN FIGURE I.3

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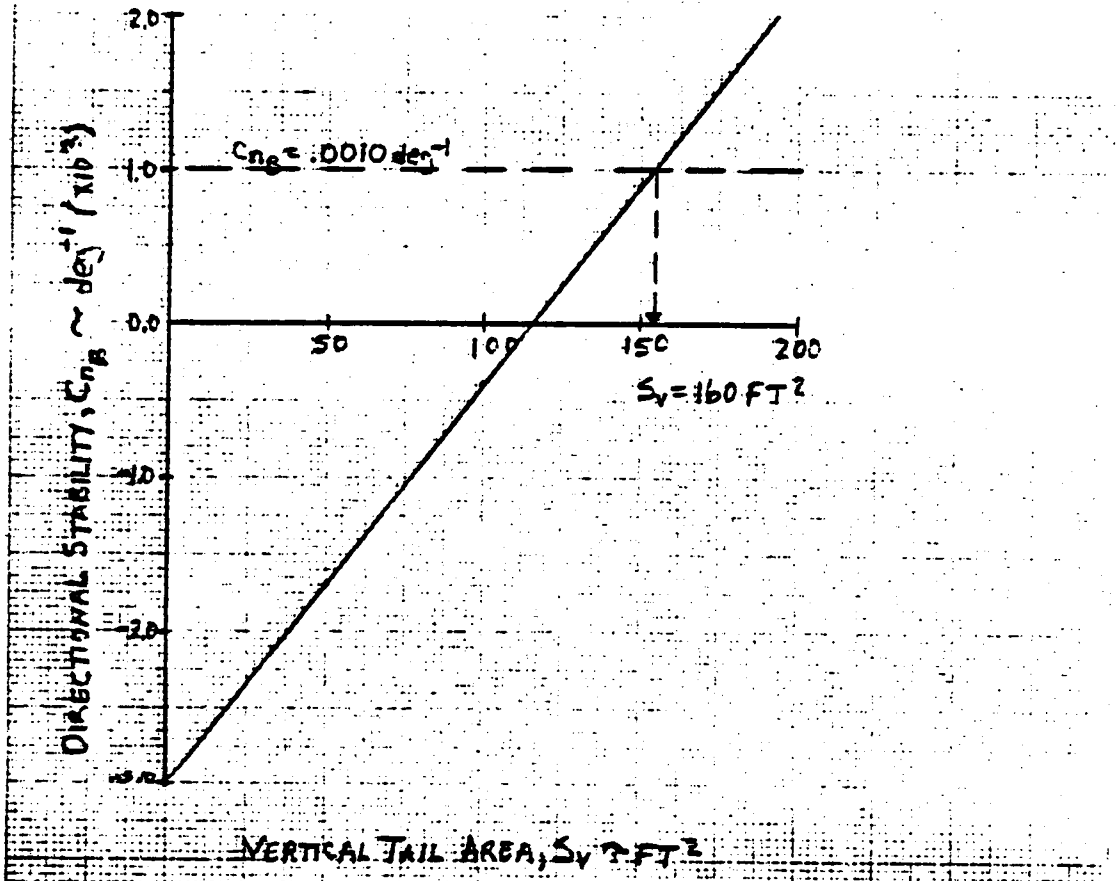

CALC	D. HENSLEY	11-02-86	REVISED	DATE	FIGURE I3: DIRECTIONAL X-PLOT
CHECK					
APPD					
APPD					
					UNIVERSITY OF KANSAS
					PAGE I-30

D Hensley | 1 Engine Out | 25 Pax | 11-12-86

$N_\delta = 0.25\, N_{t_{CRIT}} = 18183$

$V_{MC} = 1.2\, V_{S_L}$

$C_{L_{MAX_L}} = 2.1$

$V_{S_L} = 141.5$ FT/SEC

$V_{MC} = 170$ FT/SEC

$\bar{q}_{MC} = 34.4$ PSF

$$\delta_R = \frac{.0883}{C_{n_{\delta_R}}} = \frac{88.32}{S_V}$$

REQUIRES 200 FT² OF VERTICAL TAIL

FOR $y_T = 6.5$ FT

$$\delta_R = \frac{.0765}{C_{n_{\delta_R}}} = \frac{76.54}{S_V}$$

$S_V = 170$ FT² WILL YIELD

$\delta_R = 25.8$

Resized Empennage From Stability and Control

Horizontal tail

$S_H = 69\ ft^2$

$A_H = 4$

$b_H = 16.6\ ft$

$\lambda_H = .7$

$C_{r_H} = 4.9\ ft.$

$\Lambda_{LE} = 20°$

$\bar{C}_H = 4.2\ ft.$

Vertical Tail

$S_V = 170\ ft^2.$

$A_V = 1.15$

$b_V = 14\ ft.$

$\lambda_V = .3$

$C_{r_V} = 18.7\ ft.$

$C_{t_V} = 5.6\ ft.$

$\Lambda_{LE} = 54°$

$\bar{C}_V = 13.3\ ft$

H.8 Calculation of Class I Drag Polars

This section compute the airplane wetted area, and estimates skin friction drag. Class I Drag polars are constructed and compared with the polars computed from the performance sizing. Table H.9 contains a wetted area breakdown.

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List of components that contribute to wetted area:

Fuselage · Empennage

Wing Nacelles

Pylons

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1) Wing

$S_{wet} = 2S_{exp}\left[1 + .25 (t/c)_r (1+\tau\lambda)/(1+\lambda)\right]$

$\tau = 1. \qquad \lambda = .4 \qquad (t/c)_r = .13$

$S_{exp} = S_W - S_{\text{fuselage intersection}}$

$= 421 - \left[(48.3)(15) + \text{[illegible]} (48.3)(5)\right]/144$

$S_{exp} = 346 \text{ ft}^2$

$\boxed{S_{wet} = 717 \text{ ft}^2}$

3. Horizontal Tail

$S_{exp} = 69 \text{ ft}^2 \qquad \lambda = .7 \qquad \tau = 1.0 \qquad (c_t/c_r)_r = .11$

$S_{wet_H} = 2(69)\{1 + .25(.11)\}$

$\boxed{S_{wet_H} = 142 \text{ ft}^2}$

Vertical tail

$S_V = 170 \text{ ft}^2 \qquad \lambda = .3 \qquad \tau = 1.0 \qquad (c_t/c_r) = .11$

$S_{wet_V} = 2(170)[1 + .25(.11)]$

$\boxed{S_{wet_V} = 349 \text{ ft}^2}$

Fuselage

$$S_{wet_{fus}} = \pi D_f l_f (1 - 2/\lambda_f)^{2/3} (1 + 1/\lambda_f^2)$$

$$\lambda_f = l_f / D_f = 8.52$$

$$l_f = 823 \text{ in} \qquad D_f = 96.6 \text{ in.}$$

$$S_{wet_{fus}} = \pi (96.6)(823)(1 - 2/8.52)^{2/3}(1 + 1/8.52^2)/144$$

$$\boxed{S_{wet_{fus}} = 1471 \text{ ft}^2}$$

Nacelle

$$S_{wet} = \pi l_{eng} D_{eng}$$

$$S_{wet} = \pi (108.5)(38)$$

$$\boxed{S_{wet} = 90 \text{ ft}^2}$$

$$\boxed{\text{Total} = 180 \text{ ft}^2}$$

Pylons

$$S_{wet} = 2 S_{eff} [1 + .25 (t/c)_r (1 + \tau \lambda)/(1 + \lambda)]$$

$$\lambda = 1 \qquad \tau = 1.0 \qquad (t/c)_r = .11$$

$$S_{wet} = 2(70.8)[1 + .25(.11)]$$

$$\boxed{S_{wet} = 80 \text{ ft}^2}$$

TABLE H.9 Wetted Area of 25 Pax Airplane Components

Component	Wetted Area (ft²)
Wing	717
H-Tail	142
V-tail	349
Fuselage	1471
Engine Nacelles	180
Engine pylons	80
Total	2939 ft²

From Fig 3.21 of Roskam's Part I

$f = 7.2\ ft^2$ at $C_f = .0025$

$C_{D_0} = f/S = .0171$

for compressibility effects.

$\boxed{C_{D_0} = .0173}$

Take-off C_{D_0} increment:

.015 for gear

Landing C_{D_0} increment.

.015 for gear

.075 for flaps

Oswald's efficiency factor

Take-off $e = .80$

Cruise $e = .85$

Landing $e = .80$

Drag Polars

$A = 12$

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$C_D = C_{D_0} + C_L^2 / \pi A e$

Take-off $C_D = .0321 + .0332\, C_L^2$

Cruise $C_D = .0173 + .0312\, C_L^2$

Landing $C_D = .1071 + .0332\, C_L^2$

For Cruise, $C_{L_{cr}} = .30$ $(L/D)_{cr} = 14.92$

From initial weight sizing an $(L/D)_{cr} = 16$ was assumed. and $\partial W_{TO} / \partial (L/D) = -500.7$ lb. was calculated.

$\Delta L/D =$ Class I (L/D) − initial (L/D)

$= 14.92 - 16 =$

$\Delta L/D = -1.08$

Therefore: $\Delta W_{TO} = \partial W_{TO}/\partial(L/D) = (-500.7)(-1.08)$

$\Delta W_{TO} = 540.8$ lb increase.

$W_{TO\,new} = 21046 + 541 = 21587$ lb.

This represents a 2.6% change in Take-off weight. This change is so small that re-sizing of the airplane is not required.

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APPENDIX I

ENGINEERING CALCULATIONS FOR THE 36 PASSENGER COMMUTER

TABLE OF CONTENTS

I.1 INTRODUCTION

THE PURPOSE OF THIS APPENDIX IS TO PRESENT THE PRELIMINARY SIZING AND CLASS I DESIGN CALCULATIONS. METHODS USED WERE TAKEN FROM REFERENCES 1. AND 2. REFERENCES 5 AND 6. ARE USED FOR STABILITY AND CONTROL DESIGN CONSIDERATIONS.

SECTION I.2 CONTAINS PRELIMINARY WEIGHT SIZING CALCULATIONS. THESE RESULTS ARE FROM XEWTO6, A COMPUTER PROGRAM AVAILABLE AT KANSAS UNIVERSITY.

SECTION I.3 CONTAINS PRELIMINARY PERFORMANCE RESULTS FROM XPRFRM.

SECTION I.4 CONTAINS CLASS I FLAP SIZING CALCULATIONS.

SECTION I.5 CONTAINS CLASS I EMPENNAGE SIZING ($\bar{V}$-BAR METHOD)

SECTION I.6 CONTAINS LANDING GEAR DESIGN CRITERIA.

SECTION I.7 CONTAINS STABILITY AND CONTROL CALCULATIONS.

SECTION I.8 CONTAINS THE WETTED AREA CALCULATIONS AND THE CLASS I DRAG POLARS.

I.2 INITIAL WEIGHT SIZING

USING XEWTOG, A WEIGHT SIZING PROGRAM WHICH FOLLOWS THE METHOD IN CH 2. OF REFERENCE 1., THE FOLLOWING WEIGHTS AND TAKE-OFF WEIGHT SENSITIVITIES FOR THE 36 PASSENGER AIRPLANE. SEE TABLE I.1.

THE DESIGN ASSUMPTIONS USED IN THE WEIGHT SIZING ARE:

$(L/D)_{CR} = 16$

$C_p = .4$ LBS/HP/HR

$\eta_p = .85$

$V_{CR} = 442$ KTS

Table I.1 Initial Weight Sizing Results with Sensitivities

```
WEIGHT ESTIMATION FOR A REGIONAL,
PROPELLER DRIVEN AIRCRAFT
PASSENGER WEIGHT IS      7380.
CARGO WEIGHT IS            0.
CREW WEIGHT IS           615.
PHASE W/W  CJ OR CP   NP     L/D      ALTCR      RC    MCR OR V   E OR R    FLDROP
  1   0.990  0.00    0.00    0.00       0.        0.0    0.00       0.00       0.
  2   0.995  0.00    0.00    0.00       0.        0.0    0.00       0.00       0.
  3   0.995  0.00    0.00    0.00       0.        0.0    0.00       0.00       0.
  4   0.985  0.00    0.00    0.00       0.        0.0    0.00       0.00       0.
  5   0.906  0.40    0.85   16.00       0.        0.0    0.00    1100.00       0.
  7   0.985  0.00    0.00    0.00       0.        0.0    0.00       0.00       0.
  8   0.995  0.00    0.00    0.00       0.        0.0    0.00       0.00       0.
REGRESSION COEFFICIENTS ARE A=0.3959 AND B=0.9647
THE MISSION FUEL FRACTION WITHOUT RESERVES IS:0.857
THE GROSS TAKE OFF WEIGHT IS      31395. POUNDS.
THE EMPTY WEIGHT IS       17698. POUNDS.
THE WEIGHT OF FUEL IS      5620. POUNDS.



**EMPTY WEIGHT REDUCTION DUE TO COMPOSITES: 5.0 PER CENT
** SENSITIVITY ANALYSIS BEGINS HERE **
GROWTH FACTOR DUE TO PAYLOAD WEIGHT IS............          4.3
THE TAKE-OFF WEIGHT TO EMPTY WEIGHT SENSITIVITY IS          1.7



CHOICE NUMBER..        5
                    CRUISE
SFC (LB/LB/HR)            ****
SFC (LB/HP/HR)            0.40
PROP EFFICIENCY          0.85
L/D                      16.0
VELOCITY (KNOTS)           0.0
RANGE (NT. MILES)       1100.0
ENDURANCE (HRS)        *******
THE SENSITIVITY OF GROSS TAKE-OFF WEIGHT TO THE FOLLOWING PARAMETERS
IS NOW GIVEN AS THE PARTIAL DERIVATIVE OF THE GROSS TAKE-OFF WEIGHT
TO THE INDICATED PARAMETER.
DWTO/DCJ (LB/LB/LB/HR)        0.0
DWTO/DCP (LB/LB/HP/HR)    30976.4
DWTO/DNP (POUNDS)        -14577.1
DWTO/D(L/D) (POUNDS)       -774.4
DWTO/DV (LB/KNOT)             0.0
DWTO/DR (LB/NT MILE)         11.3
DWTO/DE (LB/HR)               0.0
```

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I.3 INITIAL PERFORMANCE SIZING

THE RESULTS FROM XPRFRM, A PERFORMANCE SIZING PROGRAM, ARE PRESENTED IN THIS SECTION. THE METHODS USED ARE IN CH. 3. OF REFERENCE 2. SEE TABLES I.3 THROUGH I.6.

TABLE I.2

```
******************************************************
****************  TAKE-OFF SIZING  *******************
******************************************************

           "FAR 25 CERTIFICATION CATEGORY"

                          REGIONAL TURBO-PROP

                          PROPELLER DRIVEN

------------------   INPUT DATA   ------------------

ALTITUDE                                     0.0(FEET)
FAR 23 TAKE-OFF DISTANCE <STO>               0.0 (FEET)
MINIMUM WING LOADING                       20.00(LB/FT**2)
MAXIMUM WING LOADING                      100.00(LB/FT**2)
MINIMUM TAKE-OFF LIFT COEFFICIENT 1.40
MAXIMUM TAKE-OFF LIFT COEFFICIENT 2.60

-------------------   OUTPUT DATA   ----------------

          TABLE OF POWER LOADINGS

 W/S              CLMAX-TO

0.00      1.40    1.80    2.20    2.60

 20.0     18.8    24.2    29.6    35.0
 40.0      9.4    12.1    14.8    17.5
 60.0      6.3     8.1     9.9    11.7
 80.0      4.7     6.1     7.4     8.8
100.0      3.8     4.8     5.9     7.0
```

TABLE I.3 LANDING FIELD LENGTH SIZING

```
                        REGIONAL TURBO-PROP

                   FAR 25 CERTIFICATION CATEGORY

GROSS TAKE-OFF WEIGHT  (WTO)         31395.0(LBS)
LANDING TO TAKE-OFF WEIGHT RATIO         1.000
ALTITUDE                                 0.0(FEET)
DENSITY                            .0023769(SLUG/FT**3)
LANDING APPROACH SPEED  (VA)           108.0(KTS)
LANDING FIELD LENGTH  (SFL)           3500.0(FEET)

      (W/S)TO= 23.40CLMAX(LAND)


      MAXIMUM TAKE-OFF WING LOADINGS
  TO MEET LANDING DISTANCE REQUIREMENT
=======================================

   CLMAX         MAXIMUM WING LOADING



  (LAND)              (TAKE-OFF)
                      (LB/FT**2)
 ---------       --------------------

   2.00                46.81
   2.20                51.49
   2.40                56.17
   2.60                60.85
   2.80                65.53
   3.00                70.21
```

TABLE I.4

```
*******************************************************
***************  DRAG POLAR EQUATIONS  ************
*******************************************************


         ******  INPUT DATA  ******

    MAXIMUM TAKE-OFF WEIGHT (CLEAN)     31395.0  (LBS)
    WING AREA                             500.00  (FT**2)
    ASPECT RATIO                           12.00
    SKIN FRICTION COEFFICIENT            0.00250
    AIRPLANE WETTED AREA                3393.(FT**2)

  DRAG INCREMENT DUE TO TAKE-OFF FLAPS  .0200
  DRAG INCREMENT DUE TO LANDING FLAPS   .0600
  DRAG INCREMENT DUE TO LANDING GEAR    .0150

  OSWALDS EFFICIENCY FACTOR (CLEAN)     .850
  OSWALDS EFFICIENCY FACTOR (TAKE-OFF)  .800
  OSWALDS EFFICIENCY FACTOR (LANDING)   .800

         ******  CALCULATED DATA  ******

  THE COMPLETE SET OF DRAG POLARS IS:

    1.  LOW-SPEED   (CLEAN):
    CD = .0216 + .0312CL**2         L/Dmax =19.29

    2.  TAKE-OFF   (LANDING GEAR UP):
    CD = .0416 + .0332CL**2         L/Dmax =13.47

    3.  TAKE-OFF   (LANDING GEAR DOWN):
    CD = .0566 + .0332CL**2         L/Dmax =11.55

    4.  LANDING    (LANDING GEAR UP):
    CD = .0816 + .0332CL**2         L/Dmax = 9.62




    5.  LANDING    (LANDING GEAR DOWN):
    CD = .0966 + .0332CL**2         L/Dmax = 8.84

*******************************************************
```

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TABLE I.5a)

```
FAR 25.111 (CEI)  "INITIAL CLIMB SEGMENT"
=========================================

FAR 25.111 CLIMB GRADIENT (INITIAL SEGMENT)   1.2000

               TABLE OF POWER LOADINGS REQUIRED
               --------------------------------

WING LOADING =  20.00      40.00      60.00      80.00     100.00
  (LB/FT**2)

ASPECT
```

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```
RATIO

 10.00              84.32      59.63      48.68      42.16      37.71
 11.00              88.73      62.75      51.24      44.37      39.69
 12.00              92.81      65.62      53.58      46.40      41.50
 13.00              96.54      68.26      55.74      48.27      43.17
 14.00              99.99      70.70      57.73      49.99      44.72
```

TABLE I.5 b)

```
FAR 25.121 (CEI)  "SECOND SEGMENT CLIMB"
========================================

FAR 25.121 CLIMB GRADIENT (SECOND SEGMENT)    2.4000

               TABLE OF POWER LOADINGS REQUIRED
               --------------------------------

WING LOADING =  20.00      40.00      60.00      80.00     100.00
  (LB/FT**2)

ASPECT
RATIO

 10.00              75.35      53.28      43.50      37.68      33.70
 11.00              78.87      55.77      45.53      39.43      35.27
 12.00              82.05      58.02      47.37      41.03      36.70
 13.00              84.96      60.08      49.05      42.48      37.99
 14.00              87.62      61.96      50.59      43.81      39.18
```

TABLE I.5 c)

```
FAR 25.121 (CEI)  "TRANSITION SEGMENT CLIMB"
============================================

FAR 25.121 CLIMB GRADIENT (TRANSITION)        0.1000

               TABLE OF POWER LOADINGS REQUIRED
               --------------------------------

WING LOADING =  20.00      40.00      60.00      80.00     100.00
  (LB/FT**2)

ASPECT
RATIO

 10.00              78.69      55.64      45.43      39.34      35.19
 11.00              83.84      59.28      48.40      41.92      37.49
 12.00              88.67      62.70      51.19      44.34      39.65
 13.00              93.22      65.92      53.82      46.61      41.69
 14.00              97.51      68.95      56.29      48.75      43.61
```

TABLE I5d)

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FAR 25.121 (OEI) "EN-ROUTE CLIMB SEGMENT"

FAR 25.121 CLIMB GRADIENT (EN-ROUTE) 1.2000

TABLE OF POWER LOADINGS REQUIRED

ASPECT RATIO	WING LOADING = (LB/FT**2) 20.00	40.00	60.00	80.00	100.00
10.00	90.78	64.19	52.41	45.39	40.60
11.00	94.62	66.91	54.63	47.31	42.31
12.00	98.07	69.35	56.62	49.04	43.86
13.00	101.20	71.56	58.43	50.60	45.26
14.00	104.05	73.57	60.07	52.02	46.53

TABLE I.5e)

FAR 25.119 (AEO) "LANDING CLIMB SEGMENT"

FAR 25.119 CLIMB GRADIENT (LANDING) 3.2000

TABLE OF POWER LOADINGS REQUIRED

ASPECT RATIO	WING LOADING = (LB/FT**2) 20.00	40.00	60.00	80.00	100.00
10.00	30.59	21.63	17.66	15.30	13.68
11.00	31.90	22.55	18.42	15.95	14.27
12.00	33.07	23.39	19.09	16.54	14.79
13.00	34.14	24.14	19.71	17.07	15.27
14.00	35.10	24.82	20.27	17.55	15.70

TABLE I.5f)

FAR 25.121 (OEI) "GO-AROUND OR BALKED LANDING"

FAR 25.121 CLIMB GRADIENT (GO-AROUND) 2.1000

TABLE OF POWER LOADINGS REQUIRED

ASPECT RATIO	WING LOADING = (LB/FT**2) 20.00	40.00	60.00	80.00	100.00
10.00	54.84	38.78	31.66	27.42	24.52
11.00	56.45	39.93	32.61	28.24	25.26
12.00	57.92	40.95	33.44	28.96	25.90
13.00	59.20	41.86	34.18	29.60	26.47
14.00	60.34	42.67	34.84	30.17	26.98

TABLE I.6

POWER LOADINGS NECESSARY TO MEET THE CRUISE SPEED REQUIREMENTS

(W/S) ACTUAL (psf)	(W/S) TAKEOFF (psf)	(W/P) ACTUAL (lb/hp)	(W/P) TAKEOFF IN FLIGHT (lb/hp)	(W/P) TAKEOFF STATIC (lb/hp)
18.47	20.00	1.85	2.00	1.00
36.95	40.00	3.70	4.00	2.00
55.42	60.00	5.55	6.01	3.00
73.90	80.00	7.40	8.01	4.00
92.37	100.00	9.25	10.01	5.01

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I.4 FLAP SIZING

USING A METHOD IN REF. 2. CH.7, IT WAS DETERMINED THAT THE FOLLOWING FLAP GEOMETRY WOULD SUPPLY THE INCREMENTAL LIFT NECESSARY FOR TAKE-OFF AND LANDING. SEE TABLE I.7 THE DESIGN CALCULATIONS FOLLOW.

TABLE I.7 36 PAX FLAP GEOMETRY

TRAILING EDGE FOWLER FLAPS

$C_f/c = .25$

$S_{wf}/S = .9$

$b_f/b = .9$

$\delta_f = 30°$

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CLASS I FLAP SIZING

FROM CH 7. REF 2.

$C_{L_{MAX}} = 1.4$

$C_{L_{MAX_{TO}}} = 1.4$

$C_{L_{MAX_L}} = 3.0$

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USING EQN 7.1

$C_{L_{MAX_W}} = 1.05\, C_{L_{MAX}}$

$= 1.47$

USING EQN 7.2 TO CORRECT FOR WING SWEEP:

$C_{L_{MAX_W}} = C_{L_{MAX_{W_{UNSWEPT}}}} \cos \Lambda_{c/4}$

$= 1.47 \cos 13°$

$C_{L_{MAX_W}} = 1.43$

USING EQN 7.3 USING FOR AIRFOIL SECTIONAL $C_{l_{MAX}} = 1.5$

$K_\lambda = .95$ FOR $\lambda_w = .4$

$$C_{L_{MAX_W}} = K_\lambda \frac{(C_{l_{max_r}} + C_{l_{max_t}})}{2}$$

$C_{L_{MAX_W}} = 1.42$

EQN'S 7.2 AND 7.3 $C_{L_{MAX_W}}$ RESULTS AGREE, THEREFORE TO BE CONSERVATIVE: [illegible]

$$\boxed{C_{L_{MAX_W}} = 1.42}$$

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INCREMENTAL $C_{l_{max}}$ TO BE GENERATED BY THE FLAPS:

TAKE-OFF: $\Delta C_{L_{MAX_{TO}}} = 1.05\,(C_{L_{MAX_{TO}}} - C_{L_{MAX}})$

LANDING: $\Delta C_{L_{MAX_L}} = 1.05\,(C_{L_{MAX_L}} - C_{L_{MAX}})$

$$\Delta C_{L_{MAX_{TO}}} = 0$$

$$\Delta C_{L_{MAX_L}} = 1.68$$

USING EQN 7.8

$$\Delta C_{l_{max}} = \Delta C_{L_{MAX}}\,(S/S_{WF})\,K_{\Lambda}$$

FROM EQN 7.9

$$K_{\Lambda} = .906$$

$$\Delta C_{l_{max}} = 1.52\left(\frac{S}{S_{WF}}\right)$$

FOR AN $\frac{S_{WF}}{S} = .9$ WHICH REQUIRES FULL SPAN FLAPS OVER THE WETTED WING.

$$\boxed{\Delta C_{l_{max}} = 1.69}$$

FOWLER FLAPS ARE NECESSARY TO GENERATE SUCH A HIGH $\Delta C_{l_{max}}$

FOR ΔC_l OF THE FLAP SECTION

EQN 7.11

$$\Delta C_l = (1/K)\, \Delta C_{l_{max}}$$

$K = .98$ TAKEN FROM FIGURE 7.4

$$\Delta C_l = 1.72$$

USING EQN 7.17 FOR FOWLER FLAPS

$$C_{l_{\alpha f}} = C_{l_\alpha} (1 + C_f/c)$$

$C_{l_\alpha} = 6.0$ RAD^{-1} (NLF AIRFOIL DATA)

$C_f/c = .25$

$$C_{l_{\alpha f}} = 7.5 \text{ RAD}^{-1}$$

USING EQN 7.14

$$\Delta C_l = C_{l_{\alpha f}}\, \alpha_{\delta_f}\, \delta_F$$

FOR $\delta_F = 30°$, $\alpha_{\delta_F} = .45$
FROM FIGURE 7.8

$$\Delta C_l = 7.5(.45)\left(\frac{30}{57.3}\right)$$

$$\boxed{\Delta C_l = 1.77}$$

COMPARING WITH EQN 7.11 IT IS SEEN THAT THE CHOICES FOR THE FLAPS WILL GENERATE THE REQUIRED ΔC_l.

I.5 $\bar{V}$ - BAR METHOD FOR EMPENNAGE SIZING

REF 2. CH 8.

A T-TAIL TYPE EMPENNAGE, CONVENTIONAL. TABLE I.8 CONTAINS THE GEOMETRY OF THE EMPENNAGE. TAKING AN AVERAGE VALUE FOR $\bar{V}_H$, $\bar{V}_V$ FROM TABLES 8.6a AND 8.6b. RESULTS:

$$\bar{V}_H = 1.08$$

$$\bar{V}_V = .083$$

SIMILARLY FOR S_e/S_H, S_a/S, S_r/S_V AVERAGES WERE FOUND.

$$\frac{S_e}{S_H} = .36$$

$$\frac{S_a}{S} = .06$$

$$\frac{S_r}{S_V} = .34$$

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FOR THE 36 PAK

$S = 449 \text{ FT}^2$

$\bar{c} = 6.5$ FT

$x_H = 46$ FT

$b = 73.4$

$x_V = 35$ FT

FROM EQN 8.3 REF 2.

$S_H = \bar{V}_H S \bar{c} / x_H$

$\therefore S_H = 69 \text{ FT}^2$

FROM EQN 8.4 REF 2.

$S_V = \bar{V}_V S b / x_V$

$\therefore S_V = 78 \text{ FT}^2$

FOR THE CONTROL SURFACES:

$S_a = 27 \text{ FT}^2$

$S_r = 26.5 \text{ FT}^2$

$S_e = 25 \text{ FT}^2$

TABLE I.8
GEOMETRIC DECISIONS FOR THE EMPENNAGE

HORIZONTAL TAIL	VERTICAL TAIL
$S_H = 69 \text{ FT}^2$	$S_V = 78 \text{ FT}^2$
$A_H = 4$	$A_V = 1.3$
$\lambda_H = .7$	$\lambda_V = .5$
$\Lambda_{L.E} = 20°$	$\Lambda_{LE} = 45°$
$\bar{c}_H = 4.2 \text{ FT}$	$(t/c)_V = .11$
$b_H = 16.6 \text{ FT}$	$\Gamma = 90°$
$c_r = 58.8''$	$i = 0°$
$c_t = 41''$	$b_V = 10.1 \text{ FT}$
$(t/c)_H = .11$	$c_r = 124''$
$\Gamma = 0°$	$c_t = 62''$
$i = 0°$	$\bar{c}_V = 8.04 \text{ FT}$

I.6 CLASS I LANDING GEAR DESIGN

FROM CH 9. IN REF 2. IT WAS DECIDED TO CHOOSE A 30" DIA TIRE 9" WIDE. THIS TIRE CAN CARRY 20000 LBS.

FROM WEIGHT AND BALANCE CALCULATIONS LONGITUDINAL GEAR PLACEMENT CRITERION WERE MET. THERE IS 15° BETWEEN GROUND CONTACT POINT AND AFT C.G.

FIGURE I.1. SHOWS THAT LATERAL TIP-OVER CRITERION IS MET FOR A 180" WHEEL BASE.

LATERAL TIP OVER CRITERIA TEST FOR THE 36 PAK.

TO MEET REQUIREMENT ANGLE ψ MUST BE ≤ 55°

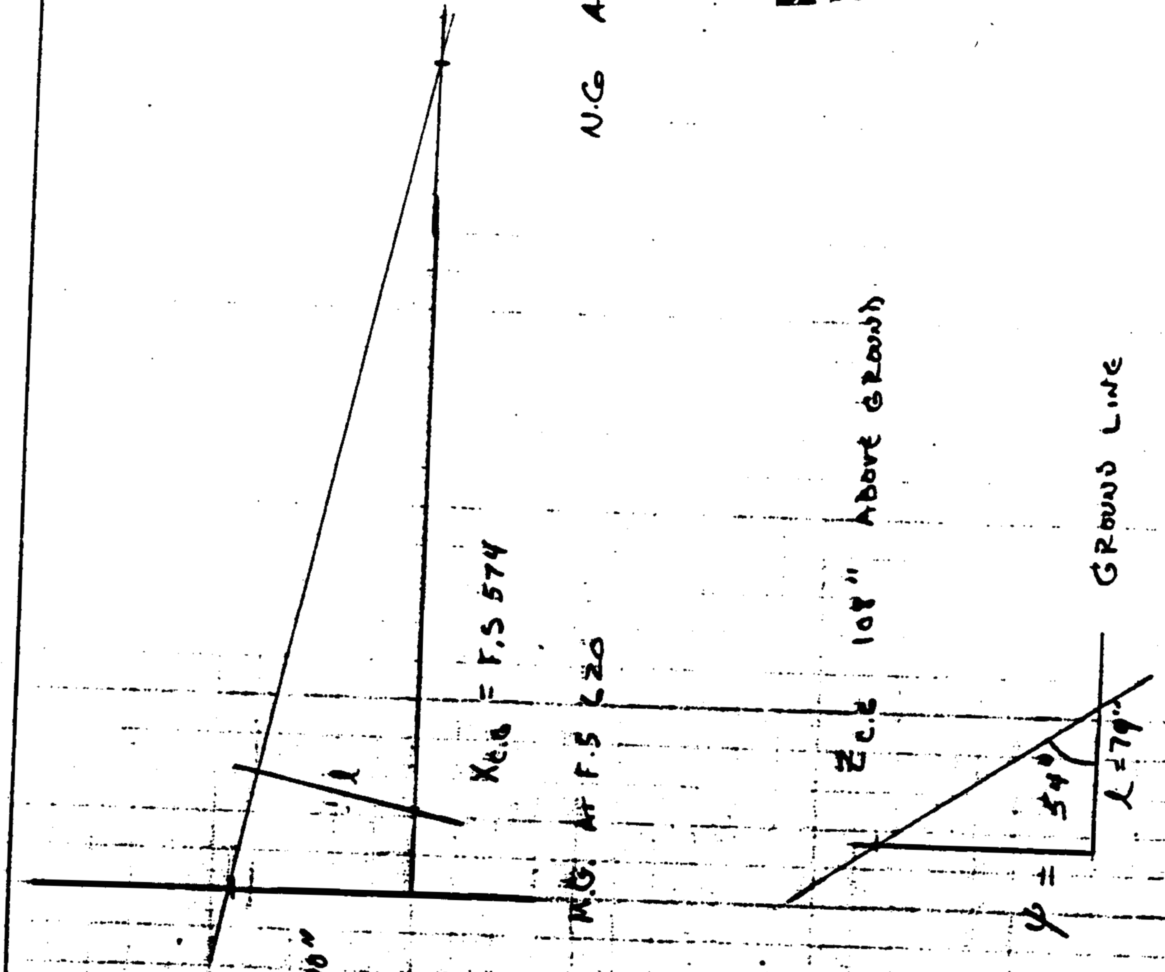

FIGURE I.1. LATERAL TIP-OVER

I.7 STABILITY AND CONTROL CALCULATIONS

CALCULATION OF REQUIRED STABILITY DERIVATIVES ARE PRESENTED IN THIS SECTION.

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36 PAX

WING LIFT CURVE SLOPE (M=.7)

$S = 449$

$\Lambda_{L.E} = 15^\circ = .2618 \text{ RAD}$

$A = 12$

FROM REF 5. FIGURE 3.12

FOR $A = 12$

$$K = 1 + [(8.2 - 2.3\Lambda_{L.E}) - (.22 - .153\Lambda_{L.E})A]/100$$

$$K = 1.0544$$

$\Lambda_{c/2} = 11.12^\circ$

$\tan \Lambda_{c/2} = .1965$

$C_{l_\alpha} = 6.0 \text{ RAD}^{-1}$ (NLF DATA)

$$\beta = \sqrt{1-M^2} = \sqrt{1-(.7)^2} = .7141$$

$$\kappa = \frac{C_{l_\alpha}}{\frac{2\pi}{\beta}} = .6820$$

$$C_{L_\alpha} = \frac{2\pi A}{\left(2 + \sqrt{\frac{A^2\beta^2}{\kappa^2}\left(1 + \frac{\tan^2\Lambda_{c/2}}{\beta^2}\right) + 4}\right)K}$$

$\boxed{C_{L_\alpha} = 4.71 \text{ RAD}^{-1}}$ AT $M = .7$

4.89 RAD^{-1} AT $M = .15$

CALCULATION OF $C_{L_{\alpha_H}}$

$\Lambda_{L.E} = 20°$, $\Lambda_{c/2} = 14.9°$

$S = 69$ FT^2

$A = 4$

$K = 1.0673$

$\tan \Lambda_{c/2} = .2669$

$C_{\ell_\alpha} = 6.0$ RAD^{-1}

$\beta^2 = .51$

$K = .6820$

$$C_{L_{\alpha_H}} = \frac{2\pi 4}{\left(2 + \sqrt{\frac{16(.51)}{(.682)^2}\left(1 + \frac{(.2669)^2}{.51}\right) + 4}\right)1.0673}$$

$$\boxed{C_{L_{\alpha_H}} = 3.41 \text{ RAD}^{-1}}$$ AT M = .7

USING REF. 5. FIG 3.12

3.46 RAD^{-1} AT M = .15

CALCULATION OF $\frac{d\epsilon}{d\alpha}$ (USING REFERENCE 5.)

FROM FIGURE 3.25

$m = 220''/440'' = .2$

$r = 428''/440'' = .973$

$\lambda = .4$

FOR $\lambda = .33$ $A = 12$

$\frac{d\epsilon}{d\alpha} = .244$

$\lambda = .20$

$\frac{d\epsilon}{d\alpha} = .250$

FOR $\lambda = .4$

$$\frac{\frac{d\epsilon}{d\alpha} - .250}{.2} = \frac{.244 - .250}{.13}$$

$\frac{d\epsilon}{d\alpha} = .241$

FROM FIGURE 3.26

$A = 12$ CORRECT $\frac{d\epsilon}{d\alpha}$ BY A FACTOR .98

$$\boxed{\frac{d\epsilon}{d\alpha} = .236}$$

$$\left(1 - \frac{d\epsilon}{d\alpha}\right) = .764$$

MULTHOP'S INTEGRATION* FOR $\Delta \bar{X}_{ACB}$ (36 PAX)

$$\frac{dM}{d\alpha} = \frac{\bar{q}}{36.5} \sum_{i=1}^{i=n} W_f^2(x_i) \left.\frac{d\bar{\varepsilon}}{d\alpha}\right|_i \Delta x_i \quad \text{DEG}^{-1}$$

	$W_f(x_i)$	Δx_i	$W_f^2(x_i)$	x_i	$\frac{d\bar{\varepsilon}}{d\alpha}$	CORRECTED $\frac{d\bar{\varepsilon}}{d\alpha}$
1	63	70	3969	401	1.03	1.06
2	88	100	7744	331	1.05	1.08
3	96	100	9216	231	1.07	1.10
4	96	100	9216	131	1.12	1.15
5	96	62	9216	31	2.10	2.16
6	96	130	9216	65		.11
7	86	100	7396	165		.29
8	62	100	3844	255		.45
9	22	70	484	335		.59
W1	40	110	1600	65		.11
W2	40	110	1600	65		.11

$C_f = 100''$

$l_H : 436''$

$\bar{c}_H = 50''$

$\bar{q} = 215.6$ PSF $= 1.497$ PSI

$C_{L_{\alpha_{WB}}} = .0822$ DEG^{-1}

$\bar{c} = 78''$

$\frac{d\varepsilon}{d\alpha} = .236$

* FROM REF 5 SECTION 3.4.6

	$w_f^2(x_i)\,\frac{d\bar{\varepsilon}}{d\alpha}\,\Delta x_i$
1	294500
2	836352
3	1013760
4	1059840
5	1234207
6	131789
7	214484
8	172980
9	19989
N1	19360
N2	19360

$$\Sigma\ 5016621$$

$$\frac{dM}{d\alpha} = \frac{\bar{q}}{36.5}(5016621)$$

$$\Delta \bar{K}_{ac_B} = \frac{-\frac{dM}{d\alpha}}{\bar{q}\,S\,\bar{c}\,C_{L_{\alpha_W}}} = \frac{\frac{5016621}{36.5}}{(449)(744)(78)(.0822)}$$

$$\boxed{\Delta \bar{K}_{ac_B} = -.33}$$

$$\bar{K}_{ac_{WB}} = \bar{K}_{ac_W} + \Delta \bar{K}_{ac_B}$$

$$\bar{K}_{ac_{WB}} = .25 - .331$$

$$\boxed{\bar{K}_{ac_{WB}} = -.081}$$

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CALCULATION FOR $\overline{X}_{AC_A}$

FROM REFERENCE 2. CHAPTER 11.

$\overline{X}_{AC_{WB}} = -.08$

$C_{L_{\alpha_H}} = 3.41 \text{ RAD}^{-1}$

$\frac{d\varepsilon_H}{d\alpha} = .236$

$\overline{X}_{AC_h} = 6.40$

$C_{L_{\alpha_{WB}}} = 4.71 \text{ RAD}^{-1}$

$$\overline{X}_{AC_A} = \frac{-.08 + \frac{3.41(.764)(.1537)\,6.40}{4.71}}{1 + \frac{3.41(.764)(.1537)}{4.71}}$$

EQN 11.1
REF. 2.

$$\boxed{\overline{X}_{AC_A} = .43}$$

FS = 605

L.E $\overline{C}_w$ = 571, F.S

$\overline{C}$ = 78"

WRITING X_{AC_A} IN TERMS OF S_H/S RESULTS IN:

$$\bar{X}_{AC_A} = \frac{-.08 + 3.54\,(S_H/S)}{1 + .553\,(S_H/S)}$$

$\bar{X}_{AC_A}$	S_H	S_H/S
.07	20	.044
.22	40	.089
.36	60	.133
.50	80	.178
.63	100	.223

PLOTTED IN FIGURE I.2

$\bar{X}_{C.G}$ SHIFT DUE TO INCREASING TAIL AREA

$\frac{W_H}{S_H} = 5.76$ LBS/FT^2 FROM CLASS I W+B

S_H	W_H	W_{EMP}	$X_{C.G}$	$\bar{X}_{C.G}$
20	115.2	565.2	592	.27
40	230.4	680.4	594	.29
60	345.6	795.6	596	.32
80	460.8	910.8	598	.35
100	576.0	1026.0	600	.37

PLOTTED IN FIGURE I.2

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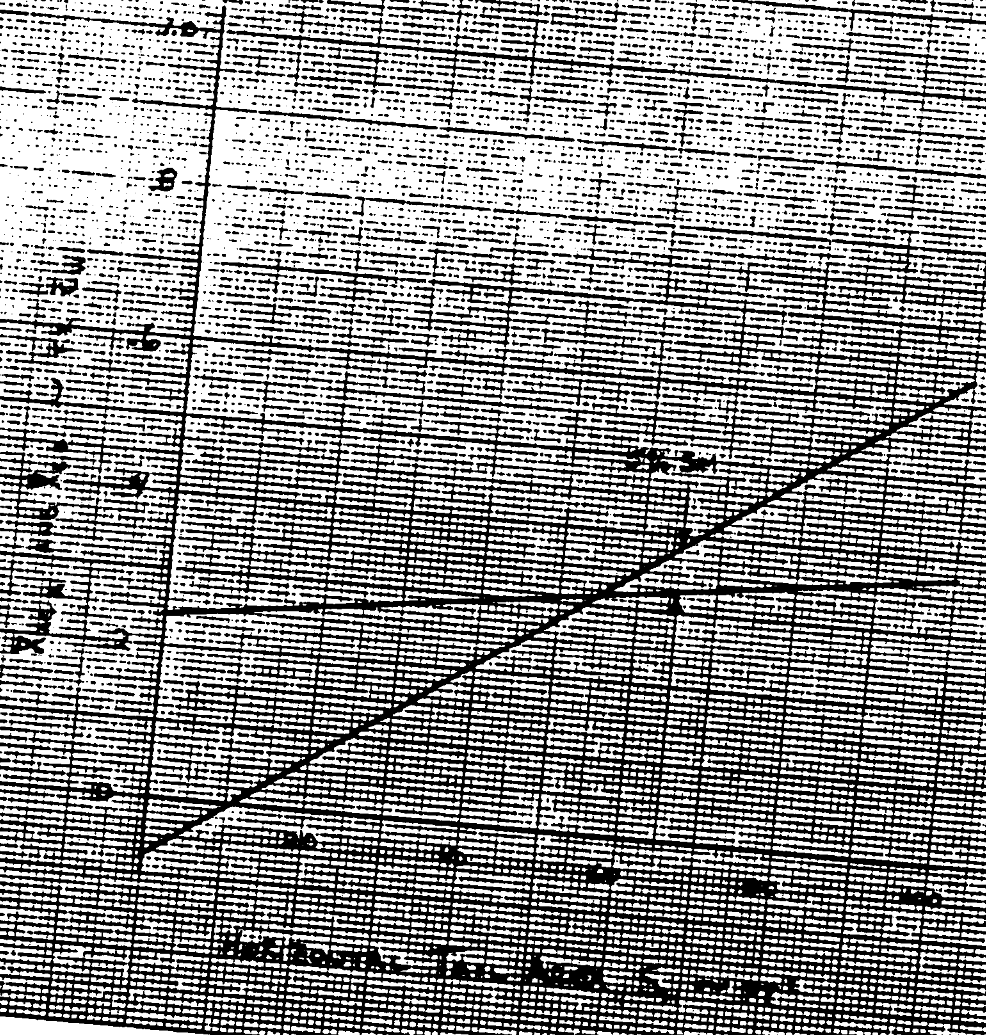

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APPD				

FIGURE II.2.–
LONGITUDINAL X-PLOT FOR
THE 36 PASSENGER COMMUTER

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CALCULATION OF $C_{L_{\alpha_V}}$

$S = 78 \text{ FT}^2$

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$\Lambda_{L.E} = 45°$

$\lambda = .5$

$X_V = 33.42$ FT , $b_w = 73.4$ FT

$A = 1.3$

$K = 1.0243$

$\tan \Lambda_{c/2} = .7436$

$\beta^2 .51$

$\kappa = .682$

$$C_{L_{\alpha_V}} = \frac{2\pi(1.3)}{\left(2 + \sqrt{\frac{(1.3)^2(.51)}{(.682)^2}\left(1 + \frac{(.7436)^2}{.51}\right) + 4}\right)1.0243}$$

$$\boxed{C_{L_{\alpha_V}} = 1.66 \text{ RAD}^{-1} \qquad M = .7}$$

FROM: REFERENCE 5. FIGURE 3.12

CALCULATION OF $C_{n_{\beta_B}}$*

$$C_{n_{\beta_B}} = -57.3\, K_N K_{R_l} \frac{S_{B_S}}{S} \frac{l_B}{b} \quad RAD^{-1}$$

$l_B = 78.1$ FT

$S = 449$ FT

$b = 73.4$ FT

$S_{B_S} = 501$ FT^2

$X_m = 41.42$ FT

$h_1 = 61''$

$h_2 = 65''$

$h = 96''$

$w = 96''$

$K_N = .001$

$$R_l = \left.\frac{V l_B}{\nu}\right|_{30000\ FT} \qquad V = 696.3\ FT/SEC \qquad \nu = 3.4926 \times 10^{-4}$$

$R_l = 1.56 \times 10^8$

$K_{R_l} = 2.03$

$$C_{n_{\beta_B}} = -57.3\,(.001)\,(2.03)\left(\frac{501}{449}\right)\left(\frac{78.1}{73.4}\right)$$

$$\boxed{C_{n_{\beta_B}} = -.138}$$

* USING METHOD IN REFERENCE 6.

CALCULATION OF $C_{n_{\beta_V}}$

$$C_{n_{\beta_V}} = C_{L_{\alpha_V}} \frac{S_V}{S} \frac{x_V}{b}$$

$$C_{n_{\beta_V}} = 1.66 \left(\frac{S_V}{449}\right)\left(\frac{33.42}{73.4}\right)$$

$$C_{n_{\beta_V}} = .0017 S_V$$

CALCULATION OF C_{n_β}

$$C_{n_\beta} = C_{n_{\beta_B}} + C_{n_{\beta_V}}$$ REFERENCE 2. EQN 11.

$$\boxed{C_{n_\beta} = -.138 + .00175 S_v}$$

S_v (FT^2)	C_{n_β} (DEG^{-1})
20	-.0018
40	-.0012
60	-.0006
80	0
100	.0006
120	.0012

$S_{V_{REQ}} = 115\ FT^2$ TO ACHIEVE $C_{n_\beta} = .001\ DEG^{-1}$

$$\boxed{\begin{array}{c} S_v = 130\ FT^2 \quad AT \quad C_{L_{\alpha_V}} = 1.46\ RAD^{-1} \\ ACHIEVES\ A\ C_{n_\beta} = .001\ DEG^{-1} \end{array}}$$

SEE FIGURE I.3

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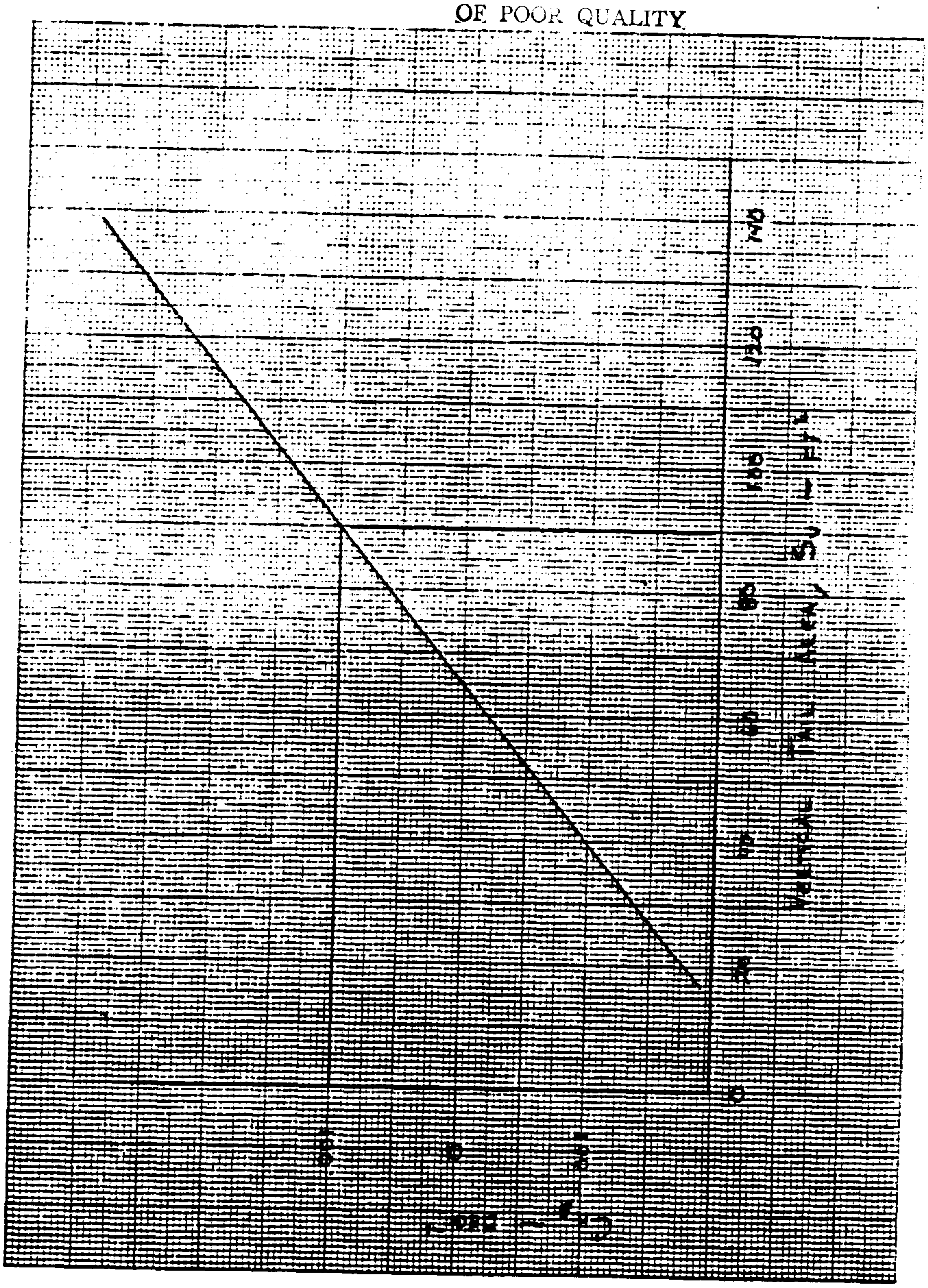

CALC	10-5-86	TRC	REVISED	DATE	FIGURE I-3 LATERAL DIRECTIONAL X-PLOT FOR THE 36 PASSENGER COMMUTER	
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ESTIMATING $C_{n_{\delta_R}}$* FOR ENGINE-OUT CALCULATION

$$C_{y_{\delta_R}} = C_{L_{\alpha_v}} \frac{(\alpha_\delta)_{C_L}}{(\alpha_\delta)_{C_\ell}} (\alpha_\delta)_{C_\ell} K' K_b \frac{S_v}{S}$$

$C_{L_{\alpha_v}} = 1.98 \text{ RAD}^{-1}$ AT $M = 0$

$(\alpha_\delta)_{C_\ell} = -.71$

$\frac{(\alpha_\delta)_{C_L}}{(\alpha_\delta)_{C_\ell}} = 1.11$

$K_b = 1$

$K' = .65$

$$C_{y_{\delta_R}} = -1.014 \frac{S_v}{S}$$

$$C_{n_{\delta_R}} = -C_{y_{\delta_R}} \frac{x_v}{b}$$

$$C_{n_{\delta_R}} = .462 \frac{S_v}{S}$$

$$\boxed{C_{n_{\delta_R}} = .0071\, S_v}$$

* FROM METHOD IN REFERENCE 6.

ENGINE - OUT CALCULATION

FROM METHOD IN REFERENCE 2. SECTION 11.3

$y_{T_{EFF}} = 6.25$ FT

$P_{TO_{REQ}} = 4485$ ONE ENGINE

$$T_{TO} = \frac{550\, P_{TO}\, \eta_p}{V_{TO}}$$

$\eta_p = .80$

$V_{TO} = 225$ FT/SEC $= 1.1\, V_{S_{TO}}$

$\therefore\ T_{TO} = 8751$ LBS

$N_{t_{CRIT}} = T_{TO}\, y_{T_{EFF}}$

$= 54694$ FT LBS

$N_\delta = .25\, N_{t_{CRIT}}$

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$V_{MC} = 1.2\, V_{S_L}$

$V_{S_L} = 140$ FT/SEC

$V_{MC} = 168$ FT/SEC

$\bar{q}_{MC} = 33.54$ PSF AT SEALEVEL

$S_R = (N_{t_{CRIT}} + N_\delta) / \bar{q}_{MC}\, S\, b\, C_{n_{\delta_R}}$ (1).

$$\boxed{S_R = 56.27 / S_V}$$

FOR $S_V = 130$ FT

$\delta_R = 24.8°$ TO HOLD ENGINE-OUT.

RESIZED VERTICAL TAIL TO HOLD ENGINE-OUT

$S_V = 130\ FT^2$

$b_V = 12\ FT$

$A_V = 1.11$

$C_t = 5\ FT$

$C_r = 16.67\ FT$

$\lambda_V = .3$

$\frac{c_r}{c_V} = .35$

$\bar{z}_V = 11.9\ FT$

I.8 CALCULATION OF CLASS I DRAG POLARS

THIS SECTION COMPUTES THE AIRPLANE WETTED AREA, AND ESTIMATES SKIN FRICTION DRAG. CLASS I DRAG POLARS ARE CONSTRUCTED AND COMPARED WITH THE POLARS COMPUTED FOR ~~THE~~ PERFORMANCE SIZING. TABLE I.9 CONTAINS A WETTED AREA BREAKDOWN.

LIST OF COMPONENTS THAT CONTRIBUTE TO WETTED AREA:

FUSELAGE

WING

EMPENNAGE

NACELLES

PYLONS

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1) WING

$$S_{WET} = 2S_{EXP\,PLF}\left[1+.25(t/c)_r(1+\tau\lambda)/(1+\lambda)\right]$$

$$S_{EXP} = \frac{b_{EXP}}{2}C_{r_{EXP}}(1+\lambda)$$

$$S_{EXP} = \frac{65.4}{2}(8.333)(1.4)$$

$$S_{EXP} = 381.5\ FT^2$$

$$2S_{EXP} = 763\ FT^2$$

$$(t/c)_r = .13$$

$$\tau = 1$$

$$\lambda = .4$$

$$S_{WET} = 2S_{EXP\,PLF}(1.0325)$$

$$\boxed{S_{WET} = 788\ FT^2}$$

Horizontal Tail

$S_{EXP} = 69$

$\lambda = .7$

$\tau = 1$

$(t/c)_r = .11$

$S_{WET} = 2(69)[1 + .25(.11)(1+.7)/(1.7)]$

$\boxed{S_{WET} = 142 \text{ FT}^2}$

Vertical Tail

$S_{EXP} = 130 \text{ FT}^2$

$\lambda = .3$

$\tau = 1$

$(t/c)_r = .11$

$S_{WET} = 2(130)[1 + .25(.11)(1+.3)/(1.3)]$

$\boxed{S_{WET} = 267 \text{ FT}^2}$

FUSELAGE

$$S_{wet_{fus}} = \pi D_f l_f (1-2/\lambda_f)^{2/3} (1+1/\lambda_f^2)$$

$\lambda_f = l_f/D_f = 9.76$

$l_f = 78.1$ FT

$D_f = 8$ FT

$$S_{wet_{fus}} = \pi(8)(78.1)(1-2/9.76)^{2/3}(1+1/(9.76)^2)$$

$$= \pi(8)(78.1)(.8582)(1.0105)$$

$$\boxed{S_{wet_{fus}} = 1702 \text{ FT}^2}$$

NACELLE

$S_{wet} = \pi\, l_{eng} D_{eng}$

$S_{wet} = \pi(108.5)(38)$

$$\boxed{S_{wet} = 90 \text{ FT}^2}$$

TOTAL ENGINE NACELLE WETTED AREA = 180 FT²

PYLONS

$$S_{wet} = 2 S_{expplf}[1+.25(t/c)_r(1+\tau\lambda)/(1+\lambda)]$$

$\lambda = 1$

$(t/c)_r = .12$

$\tau = 1$

$S_{wet} = 2(5)(6)[1.03]$

$S_{wet} = 62$ FT²

2 PYLONS $= \boxed{124 \text{ FT}^2 = S_{wet}}$

TABLE I.9 WETTED AREA OF 36 PASSENGER AIRPLANE COMPONENTS

COMPONENT	WETTED AREA (FT^2)
WING	788
H-TAIL	142
V-TAIL	267
FUSELAGE	1702
ENGINE NACELLES	90 x 2
ENGINE PYLONS	62 x 2
TOTAL	3203

FROM FIGURE 3.21 ROSKAM'S PART I

$f = 7.8\ FT^2$

$C_{D_0} = f/S = .0174$

ADD .0002 TO C_{D_0} FOR COMPRESSIBILITY EFFECTS

$\boxed{C_{D_0} = .0176}$

TAKE-OFF C_{D_0} INCREMENT

.015 FOR GEAR

LANDING C_{D_0} INCREMENT

.015 FOR GEAR
.075 FOR FLAPS

DRAG POLARS

TAKE-OFF

$C_{D_0} = .0174 + .015$

$= .0324$

$e = .80$

$A = 12$

$$C_D = C_{D_0} + \frac{C_L^2}{\pi A e}$$

$$\boxed{C_D = .0324 + .0332\, C_L^2}$$

CRUISE

$C_{D_0} = .0176$

$e = .85$

$A = 12$

$$\boxed{C_D = .0176 + .0312\, C_L^2}$$

LANDING

$C_{D_0} = .0174 + .09$

$= .1074$

$e = .8$

$A = 12$

$$\boxed{C_D = .1074 + .0332\, C_L^2}$$

$C_{L_{CRUISE}} = .3$

$(L/D)_{CR} = 14.92$

BY TAKING A 10% REDUCTION IN C_{D_0}

$.0173(.9) = .0156$

$C_D = .0156 + .0312\,C_L^2$

$$\boxed{(L/D)_{CR} = 16.32}$$

UTILIZING NLF SHOULD ALLOW A 10% REDUCTION IN AIRPLANE C_{D_0}.

FROM INITIAL WEIGHT SIZING AN $(L/D)_{CR} = 16$ WAS ASSUMED

AND: $\dfrac{\partial W_{TO}}{\partial L/D} = -774.4$ WAS CALCULATED.

$\Delta L/D$ = CLASS I L/D − INITIAL L/D

$= 14.92 - 16$

$= -1.08$

THEREFORE

$\Delta W_{TO} = \dfrac{\partial W_{TO}}{\partial L/D}\,\Delta L/D$

$\Delta W_{TO} = 836$ LBS INCREASE END

$$W_{TO_{NEW}} = W_{TO} + \Delta W_{TO}$$

$$W_{TO_{NEW}} = 3,395 + 836$$

$$= 32231 \text{ LBS}$$

THIS REPRESENTS AN 2.7 % CHANGE IN TAKE-OFF WEIGHT. THIS MAGNITUDE OF CHANGE DOES NOT WARRANT RESIZING.

APPENDIX J

ENGINEERING CALCULATIONS FOR THE 50 PASSENGER COMMUTER

TABLE OF CONTENTS

J.1 INTRODUCTION

The purpose of this appendix is to present the preliminary sizing and Class I design calculations. Methods used were taken from References 1 and 2. References 5 and 6 are used for stability and control design considerations.

Section J.2 contains preliminary weight sizing calculations. These results are from XEWTOG, a computer program available at the University of Kansas.

Section J.3 contains preliminary performance results from XPRFRM, another computer program at K.U.

Section J.4 contains Class I flap sizing calculations.

Section J.5 contains Class I empennage sizing ($\overline{V}$-bar method).

Section J.6 contains landing gear design criteria.

Section J.7 contains stability and control calculations.

Section J.8 contains the wetted area calculations and the Class I drag polars.

J.2 INITIAL WEIGHT SIZING

Using XEWTOG, a weight sizing program which follows the method in Chapter 2 of Reference 1, the following weights and take-off weight sensitivities are determined for the 50 passenger airplane. See Table J.1.

The design assumptions used in the weight sizing are:

$(L/D)_{CR} = 16$

$C_P = 0.4$ lbs/hp/hr

$\eta_p = 0.85$

$V_{CR} = 442$ kts

TABLE J.1 INITIAL WEIGHT SIZING RESULTS WITH SENSITIVITIES

```
WEIGHT ESTIMATION FOR A REGIONAL.
PROPELLER DRIVEN AIRCRAFT
PASSENGER WEIGHT IS    10250.
CARGO WEIGHT IS        0.
CREW WEIGHT IS       615.
PHASE W/W CJ OR CP   NP   L/D     ALTCR     RC   MCR OR V  E OR R   PLDROP
 1   0.990  0.00  0.00  0.00      0.       0.0     0.00      0.00       0.
 2   0.995  0.00  0.00  0.00      0.       0.0     0.00      0.00       0.
 3   0.995  0.00  0.00  0.00      0.       0.0     0.00      0.00       0.
  4   0.994  0.50  0.77  16.00  30000.  3000.0   270.00      0.00       0.
  5   0.909  0.40  0.85  16.00      0.      0.0     0.00   1055.00       0.
  7   0.985  0.00  0.00   0.00      0.      0.0     0.00      0.00       0.
  8   0.995  0.00  0.00   0.00      0.      0.0     0.00      0.00       0.
 REGRESSION COEFFICIENTS ARE A=0.3989 AND B=0.9647
 THE MISSION FUEL FRACTION WITHOUT RESERVES IS:0.868
 THE GROSS TAKE OFF WEIGHT IS     42057. POUNDS.
 THE EMPTY WEIGHT IS     23963. POUNDS.
 THE WEIGHT OF FUEL IS      6913. POUNDS.



^?^?EMPTY WEIGHT REDUCTION DUE TO COMPOSITES: 5.0 PER CENT

** SENSITIVITY ANALYSIS BEGINS HERE **
GROWTH FACTOR DUE TO PAYLOAD WEIGHT IS............        4.2
THE TAKE-OFF WEIGHT TO EMPTY WEIGHT SENSITIVITY IS        1.7



CHOICE NUMBER..       4                 5
               CLIMB TO CRUISE    CRUISE
SFC (LB/LB/HR)              ****          ****
SFC (LB/HP/HR)               0.50          0.40
PROP EFFICIENCY             0.77          0.85
L/D                         16.0          16.0
VELOCITY (KNOTS)            270.0           0.0
RANGE (NT. MILES)         *******        1055.0
ENDURANCE (HRS)           *******       *******
THE SENSITIVITY OF GROSS TAKE-OFF WEIGHT TO THE FOLLOWING PARAMETERS
IS NOW GIVEN AS THE PARTIAL DERIVATIVE OF THE GROSS TAKE-OFF WEIGHT
TO THE INDICATED PARAMETER.
DWTO/DCJ (LB/LB/LB/HR)        0.0            0.0
DWTO/DCP (LB/LB/HP/HR)        0.0        39784.5
DWTO/DNP (POUNDS)             0.0       -18722.1
DWTO/D(L/D) (POUNDS)          0.0         -994.6

DWTO/DV (LB/KNOT)             0.0            0.0
DWTO/DR (LB/NT MILE)          0.0           15.1
DWTO/DE (LB/HR)            5619.8            0.0
```

J.3 INITIAL PERFORMANCE SIZING

The results from XPRFRM, a performance sizing program, are given in this section. The methods used are in Ch. 3 of Reference 2. See Tables J.3 through J.6.

TABLE J.2 TAKE-OFF SIZING

```
**************************************************
**************   TAKE-OFF SIZING   ***************
**************************************************

          "FAR 25 CERTIFICATION CATEGORY"

                     REGIONAL TURBO-PROP

                  PROPELLER DRIVEN

------------------  INPUT DATA  ------------------

ALTITUDE                                  0.0(FEET)
FAR 25 TAKE-OFF DISTANCE <STO>            0.0 (FEET)
MINIMUM WING LOADING                    20.00(LB/FT**2)
MAXIMUM WING LOADING                   100.00(LB/FT**2)
MINIMUM TAKE-OFF LIFT COEFFICIENT 1.40
MAXIMUM TAKE-OFF LIFT COEFFICIENT 2.20

-------------------  OUTPUT DATA  ----------------

          TABLE OF POWER LOADINGS

 W/S           CLMAX-TO

0.00      1.40    1.60    1.80    2.00    2.20

 20.0     18.8    21.5    24.2    26.9    29.6
 40.0      9.4    10.8    12.1    13.4    14.8
 60.0      6.3     7.2     8.1     9.0     9.9
 80.0      4.7     5.4     6.1     6.7     7.4
100.0      3.8     4.3     4.8     5.4     5.9
```

TABLE J.3 LANDING FIELD LENGTH SIZING

```
                     REGIONAL TURBO-PROP

                 FAR 25 CERTIFICATION CATEGORY

GROSS TAKE-OFF WEIGHT  (WTO)       42057.0(LBS)
LANDING TO TAKE-OFF WEIGHT RATIO       1.000
ALTITUDE                               0.0(FEET)
DENSITY                            .0023769(SLUG/FT**3)
LANDING APPROACH SPEED  (VA)         108.0(KTS)
LANDING FIELD LENGTH  (SFL)         3500.0(FEET)

        (W/S)TO= 23.40CLMAX(LAND)

        MAXIMUM TAKE-OFF WING LOADINGS
      TO MEET LANDING DISTANCE REQUIREMENT
    ======================================

     CLMAX         MAXIMUM WING LOADING

     (LAND)             (TAKE-OFF)
                        (LB/FT**2)
    --------      -------------------

     2.20               51.49
     2.40               56.17
     2.60               60.85
     2.80               65.53
     3.00               70.21
```

TABLE J.4 INITIAL DRAG POLARS

```
*************************************************
**************  DRAG POLAR EQUATIONS  ************
*************************************************


           ******  INPUT DATA  ******

     MAXIMUM TAKE-OFF WEIGHT (CLEAN)     42057.0  (LBS)
     WING AREA                             510.00  (FT**2)
     ASPECT RATIO                           12.00
     SKIN FRICTION COEFFICIENT            0.00250
     AIRPLANE WETTED AREA                4716.(FT**2)

  DRAG INCREMENT DUE TO TAKE-OFF FLAPS  .0150
  DRAG INCREMENT DUE TO LANDING FLAPS   .0300
  DRAG INCREMENT DUE TO LANDING GEAR    .0200

  OSWALDS EFFICIENCY FACTOR (CLEAN)     .850
  OSWALDS EFFICIENCY FACTOR (TAKE-OFF)  .800
  OSWALDS EFFICIENCY FACTOR (LANDING)   .800

           ******  CALCULATED DATA  ******

  THE COMPLETE SET OF DRAG POLARS IS:

    1.  LOW-SPEED   (CLEAN):
    CD = .0284 + .0312CL**2          L/Dmax =16.79

    2.  TAKE-OFF   (LANDING GEAR UP):
    CD = .0434 + .0332CL**2          L/Dmax =13.19

    3.  TAKE-OFF   (LANDING GEAR DOWN):
    CD = .0634 + .0332CL**2          L/Dmax =10.90

    4.  LANDING    (LANDING GEAR UP):
    CD = .0584 + .0332CL**2          L/Dmax =11.36

    5.  LANDING    (LANDING GEAR DOWN):
    CD = .0784 + .0332CL**2          L/Dmax = 9.81

*************************************************
```

TABLE J.5 a)

FAR 25.111 (OEI) "INITIAL CLIMB SEGMENT"

FAR 25.111 CLIMB GRADIENT (INITIAL SEGMENT) 1.2000

TABLE OF POWER LOADINGS REQUIRED

WING LOADING = (LB/FT**2) ASPECT RATIO	20.00	40.00	60.00	80.00	100.00
10.00	80.28	56.74	46.33	40.12	35.89
11.00	84.03	59.42	48.51	42.01	37.58
12.00	87.45	61.85	50.50	43.73	39.11
13.00	90.60	64.06	52.31	45.30	40.52
14.00	93.47	66.09	53.96	46.73	41.80

TABLE J.5 b)

FAR 25.121 (OEI) "SECOND SEGMENT CLIMB"

FAR 25.121 CLIMB GRADIENT (SECOND SEGMENT) 2.4000

TABLE OF POWER LOADINGS REQUIRED

WING LOADING = (LB/FT**2) ASPECT RATIO	20.00	40.00	60.00	80.00	100.00
10.00	71.69	50.69	41.39	35.84	32.06
11.00	74.69	52.81	43.12	37.34	33.40
12.00	77.39	54.72	44.68	38.69	34.61
13.00	79.88	56.45	46.09	39.92	35.70
14.00	82.05	58.02	47.37	41.03	36.69

TABLE J.5 c)

FAR 25.121 (OEI) "TRANSITION SEGMENT CLIMB"

FAR 25.121 CLIMB GRADIENT (TRANSITION) 0.0010

TABLE OF POWER LOADINGS REQUIRED

WING LOADING = (LB/FT**2) ASPECT RATIO	20.00	40.00	60.00	80.00	100.00
10.00	70.04	49.52	40.44	35.02	31.32
11.00	74.32	52.55	42.91	37.16	33.24
12.00	78.31	55.37	45.21	39.15	35.02
13.00	82.03	58.01	47.36	41.02	36.69
14.00	85.52	60.47	49.38	42.76	38.25

TABLE J.5 d)

```
FAR 25.121 (OEI)  "EN-ROUTE CLIMB SEGMENT"
==========================================

FAR 25.121 CLIMB GRADIENT (EN-ROUTE)                  1.2000

                 TABLE OF POWER LOADINGS REQUIRED
                 --------------------------------

WING LOADING =  20.00      40.00      60.00      80.00     100.00
   (LB/FT**2)

ASPECT
RATIO

10.00              85.36      62.37      49.29      43.67      38.16
11.00              88.29      64.05      51.32      44.44      39.73
12.00              92.24      65.05      53.14      46.02      41.16
13.00              94.89      67.10      54.76      47.44      42.45
14.00              97.47      68.92      56.25      48.74      43.59
```

TABLE J.5 e)

```
FAR 25.119 (AEO)  "LANDING CLIMB SEGMENT"
=========================================

FAR 25.119 CLIMB GRADIENT (LANDING)                   3.2000

                 TABLE OF POWER LOADINGS REQUIRED
                 --------------------------------

WING LOADING =  20.00      40.00      60.00      80.00     100.00
   (LB/FT**2)

ASPECT
RATIO

 10.00             31.02      21.93      17.91      15.51      13.87
 11.00             32.26      22.81      18.63      16.13      14.43
 12.00             33.38      23.60      19.27      16.69      14.93
 13.00             34.39      24.31      19.85      17.19      15.38
 14.00             35.30      24.96      20.38      17.65      15.79
```

TABLE J.6 f)

```
FAR 25.121 (OEI)  "GO-AROUND OR BALKED LANDING"
===============================================

FAR 25.121 CLIMB GRADIENT (GO-AROUND)                 2.1000

                 TABLE OF POWER LOADINGS REQUIRED
                 --------------------------------

WING LOADING =  20.00      40.00      60.00      80.00     100.00
   (LB/FT**2)

ASPECT
RATIO

 10.00             51.84      36.66      29.93      25.92      23.15
 11.00             53.21      37.63      30.72      26.60      23.80
 12.00             54.41      38.47      31.41      27.20      24.35
 13.00             55.46      39.22      32.02      27.73      24.80
 14.00             56.40      39.88      32.56      28.20      25.22
```

TABLE J.6

```
                     POWER LOADINGS NECESSARY
              TO MEET THE CRUISE SPEED REQUIREMENTS
=================================================================

  (W/S)       (W/S)       (W/P)         (W/P)          (W/P)
 ACTUAL      TAKEOFF     ACTUAL        TAKEOFF        TAKEOFF
                                      IN FLIGHT       STATIC
  (psf)       (psf)     (lb/hp)        (lb/hp)        (lb/hp)
-----------------------------------------------------------------

    17.00       20.00       2.09          2.45           1.23
    34.00       40.00       4.17          4.91           2.45
    51.00       60.00       6.26          7.36           3.68
    68.00       80.00       8.34          9.82           4.91
    85.00      100.00      10.43         12.27           6.14
```

J.4 FLAP SIZING

Using the method of Ref. 2, Ch. 7, it is determined that the following flap geometry will supply the incremental lift necessary for take-off and landing. See Table J.7. The design calculations follow.

TABLE J.7 50 PAX FLAP GEOMETRY

TYPE: TRAILING EDGE FOWLER FLAPS

$C_f/C = 0.25$

$S_{wf}/S = 0.682$

$b_f/b = 0.638$

TAKE-OFF $\delta_F = 20$ deg

LANDING $\delta_F = 40$ deg

$\eta_i = 0.10$

$\eta_o = 0.738$

CLASS I FLAP SIZING - from Ch. 7, Ref. 2

$C_{L_{max}} = 1.5$

$C_{L_{max_{TO}}} = 2.0$

$C_{L_{max_L}} = 3.0$

$R_{N_r} = \rho V C_r / \mu = 1.28 \times 10^7$

$R_{N_t} = \rho V C_t / \mu = 5.11 \times 10^6$

From Fig. 7 of Ref. 2, a .13 airfoil yields

$C_{\ell_{max_{root}}} = 1.85$

$C_{\ell_{max_{tip}}} = 1.7$

With Eqn. (7.3)

$C_{L_{max_w}} = .95(1.85+1.7)/2 = 1.69$

Using Eqn. (7.2) $C_{L_{max_w}} = C_{L_{max_{unswept}}} \cos \Lambda_{c/4}$

$= 1.69 \cos 13.1$

$C_{L_{max_w}} = 1.65$

with Eqn (7.1) this yields

$C_{L_{max}} = 1.65/1.08 = 1.53$

Therefore, the assumed value of $C_{L_{max}} = 1.5$ is reasonable.

Incemental $C_{L_{max}}$ values to be generated by the flaps:

Take-off: $\Delta C_{L_{max_{TO}}} = 1.05(C_{L_{max_{TO}}} - C_{L_{max}})$

$\Delta C_{L_{max_L}} = 1.05(C_{L_{max_L}} - C_{L_{max}})$

$\Delta C_{L_{max_{TO}}} = 0.525$

$\Delta C_{L_{max_L}} = 1.58$

From Eqn. (7.9) $K_\Lambda = 0.906$

Using Eqn. (7.8):

T.O.: $\Delta C_{\ell_{max}} = .476\ (S/S_{WF})$

LAND: $\Delta C_{\ell_{max}} = 1.43\ (S/S_{WF})$

Assuming Fowler flaps with

$c_f/c = 0.25 \quad \delta_{F_{TO}} = 20^\circ \quad \delta_{F_L} = 40^\circ$

From Fig. 7.4, $K = .96$ and with Eqn. (7.11):

$\Delta C_\ell = (1/K)\,\Delta C_{\ell_{max}}$

T.O.: $\Delta C_\ell = .496\ (S/S_{WF})$

LAND: $\Delta C_\ell = 1.49\ (S/S_{WF})$

Obtainable $\Delta C_{\ell_{FLAP}}$:

Using Eqn. (7.17), $C_{\ell_{\alpha_f}} = 7.85$

Since $C_{L_{max_L}}$ is more critical only it is considered.

From Fig. 7.8 $\alpha_{\delta_F} = 0.40$

With Eqn. (7.14), $\Delta C_{\ell_{max_L}} = 2.10$

Thus this will generate the required $C_{\ell_{max}}$.

J.5 V-BAR METHOD FOR EMPENNAGE SIZING

- Reference 2 Chapter 8

Conventional T-Tail type empennage

Taking an average value for $\bar{V}_H$, $\bar{V}_V$ from Tables 8.6a and 8.6 b:

$$\bar{V}_H = 1.08$$

$$\bar{V}_V = .083$$

Similar averages are found for the following:

$$\frac{S_e}{S_H} = .36$$

$$\frac{S_a}{S} = .06$$

$$\frac{S_r}{S_v} = .34$$

For the 50 pax:

$S = 592 \text{ ft}^2$

$\bar{c} = 7.46 \text{ ft}$

$X_H = 36.7 \text{ ft}$

$b = 84.3 \text{ ft}$

$X_V = 31.9 \text{ ft}$

From Eqn. 8.3 Ref. 2

$$S_H = \bar{V}_H S \bar{c} / X_H$$

$$\therefore \quad S_H = 130 \text{ ft}^2$$

From Eqn. (8.4),

$$S_V = \bar{V}_V S b / x_V$$

$$S_V = 130 \text{ ft}^2$$

For the control surfaces

$$S_a = 35.5 \text{ ft}^2$$

$$S_r = 44.2 \text{ ft}^2$$

$$S_e = 46.8 \text{ ft}$$

Table J.8 Geometric Quantities for the Empennage

Horizontal Tail	Vertical Tail
$S_H = 130 \text{ ft}^2$	$S_V = 130 \text{ ft}^2$
$A_H = 5$	$A_V = 1.4$
$\lambda_H = .5$	$\lambda_V = .5$
$\Lambda_{LE} = 25°$	$\Lambda_{LE} = 40°$
$\bar{c}_H = 5.29 \text{ ft}$	$\bar{c}_V = 9.96 \text{ ft}$
$b_H = 25.5 \text{ ft}$	$b_V = 13.5 \text{ ft}$
$C_r = 6.80 \text{ ft}$	$C_r = 12.8 \text{ ft}$
$C_t = 3.40 \text{ ft}$	$C_t = 6.42 \text{ ft}$
$(t/c)_H = .12$ root, $.10$ tip	$(t/c)_V = .13$ root, $.12$ tip
$\Gamma = 0°$	$\Gamma = 90°$
$i = 0°$	$i = 0°$

J.6 CLASS I LANDING GEAR DESIGN

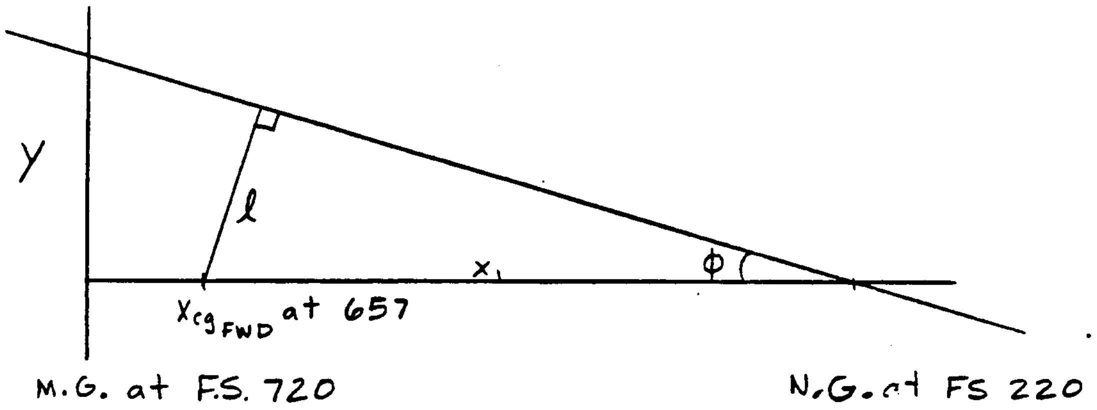
From Ch.9 of Ref. 2, it is decided to choose a 30 inch tire diameter with a width of 9 inches. This tire can carry 20000 lbs.

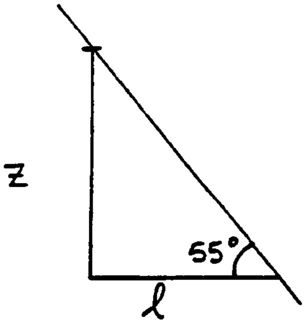
From weight and balance calculations longitudinal gear placement criteria were met. The is a 21° angle between the ground contact point and the aft c.g.

Fig. J.1 shows that the lateral tip-over criterion is met for a 198 inch wheel base.

LATERAL TIP OVER CRITERION TEST FOR THE 50 PAX AIRPLANE

To meet the requirement the angle ψ must be $\leq 55°$.





Z = 121 in.

$\tan 55° = \frac{121}{l}$ $l = 84.7$ in

$X_1 = 657 - 220 = 437$ in

$\phi = \sin^{-1}\frac{l}{X_1} = \sin^{-1}\frac{84.7}{437} = 11.2°$

$y = (720 - 220) \tan 11.2°$

$y = 98.8$ in

Thus, wheel base = 198 in

FIGURE J.1 LATERAL TIP-OVER CRITERION

J.7 STABILITY AND CONTROL CALCULATIONS

The calculation of required stability derivatives are presented in this section.

$C_{L_{\alpha_W}}$, Page J.21

$C_{L_{\alpha_H}}$, Page

$d\varepsilon/d\alpha$, Page

Multhopp Integration, Page

$\overline{X}_{AC_A}$, Page

$C_{L_{\alpha_V}}$, Page

$C_{n_{\beta_E}}$, Page

$C_{n_{\beta_V}}$, Page

C_{n_β}, Page

$C_{n_{\delta_R}}$, Page

V_{MC}, OEI., Page

50 PAX

WING LIFT-CURVE SLOPE

$S = 592$
$\Lambda_{LE} = 15° = .2618$ RAD
$A = 12$

From Ref. 5, Fig. 3.12
For A=12,

$$K = 1 + [(8.2 - 2.3\Lambda_{LE}) - (2.2 - .153\Lambda_{LE})A]/100$$

$$K = 1.054$$

$$\Lambda_{c/2} = 11.1°$$

$C_{\ell_\alpha} = 6.0 \text{ RAD}^{-1}$ (from NLF data)

$$\beta = \sqrt{1-M^2} = \sqrt{1-(.7)^2} = .714$$

$$\kappa = \frac{C_{\ell_\alpha}}{2\pi/\beta} = .682$$

Using the Polhamus Eqn. from Ch. 3 of Ref. 5:

$$C_{L_\alpha} = \frac{A}{K}\,\frac{2\pi}{\left(2 + \sqrt{\frac{A^2\beta^2}{\kappa^2}\left(1 + \frac{\tan^2\Lambda_{c/2}}{\beta^2}\right) + 4}\right)}$$

$$\boxed{C_{L_\alpha} = 4.72 \text{ RAD}^{-1}} \quad M = .70$$

At $M = 0$ $\quad C_{L_\alpha} = 4.79 \text{ RAD}^{-1}$

Calculation of $C_{L_{\alpha_H}}$

$A = 5$ $\Lambda_{c/2} = 18.4°$ $\Lambda_{LE} = 25°$

$\lambda = .50$

$K = 1.064$

At $M = .70$ $\beta = .714$ $\varkappa = .683$

At $M = 0$ $\beta = 1.0$ $\varkappa = .957$

Using the Polhamus equation again:

At $M = .70$ $C_{L_{\alpha_H}} = 3.64 \text{ RAD}^{-1}$

At $M = 0$ $C_{L_{\alpha_H}} = 3.75 \text{ RAD}^{-1}$

Calculation of $\frac{d\varepsilon}{d\alpha}$ (using Ref. 5)

From Fig. 3.25

$m = \frac{230}{565} = .407$ $\quad\quad l_h/c_r = 3.75$

$r = \frac{640}{565} = 1.13$

$\lambda = .4$

This yields $\boxed{\frac{d\varepsilon}{d\alpha} = .325}$

$1 - \frac{d\varepsilon}{d\alpha} = .675$

Multhopp Integration* for $\Delta \bar{X}_{AC_B}$ (50 PAX)

$$\frac{dM}{d\alpha} = \frac{\bar{q}}{36.5} \sum_{i=1}^{n} w_f^2(x_i) \left.\frac{d\bar{\varepsilon}}{d\alpha}\right|_i \Delta x_i \qquad DEG^{-1}$$

	Δx_i	$w_f(x_i)$	x_i	$d\bar{\varepsilon}/d\alpha$	CORRECTED $d\bar{\varepsilon}/d\alpha$
1	120	72	422	1.0	1.03
2	100	97	325	1.02	1.05
3	100	97	225	1.03	1.06
4	100	97	125	1.15	1.18
5	78	97	39	1.90	1.96
6	150	97	75		.115
7	150	97	225		.344
8	205	80	382		.583
N	110	158	112		.171

C_f = 113 in

l_H = 442 in

$\bar{c}_H$ = 5.29 ft = 63.5 in

$\bar{q}$ = 216 psf

$\frac{d\varepsilon}{d\alpha}$ = .325

* From Ref. 5 section 3.4.6

	$W_f^2(x_i)$	$\frac{d\bar{\epsilon}}{d\alpha}$	Δx_i
1	641	×	$\frac{10^3}{12^3}$
2	988		"
3	997		"
4	1110		"
5	1438		"
6	162		"
7	486		"
8	765		"
N	470		"

$\Sigma = 4084$

$$\frac{dM}{d\alpha} = 111.9\ \bar{q} \quad DEG^{-1}$$

$$\Delta \bar{X}_{AC_B} = \frac{-dM/d\alpha}{\bar{q} S \bar{c} C_{L_{\alpha W}}} = \frac{-111.9\ \bar{q}}{\bar{q}(592)(7.46)(.0823)}$$

$$\boxed{\Delta \bar{X}_{AC_B} = -.308}$$

$$\bar{X}_{AC_{WB}} = \bar{X}_{AC_W} + \Delta \bar{X}_{AC_{WB}}$$

$$= .25 - .308$$

$$\boxed{\bar{X}_{AC_{WB}} = -.058}$$

CALCULATION OF $\bar{X}_{AC_A}$

From Ref. 2, Ch. 11.

$\bar{X}_{AC_{WB}} = -.058$

$C_{L_{\alpha_H}} = 3.64 \ RAD^{-1}$

$\frac{d\varepsilon}{d\alpha} = .325$

$\bar{X}_{AC_H} = 6.38$

$C_{L_{\alpha_{wb}}} = 4.72 \ RAD^{-1}$

From Eqn. 11.1 of Ref. 2

$$\bar{X}_{AC_A} = \frac{-.058 + 3.64(1-.325)(S_H/S)(6.38)/4.72}{1 + 3.64(1-.325)(S_H/S)/4.72}$$

$$\bar{X}_{AC_A} = \frac{-.058 + 3.32(S_H/S)}{1 + .521(S_H/S)}$$

S_H	S_H/S	$\bar{X}_{AC_A}$
50	.0845	.213
100	.1689	.462
150	.2534	.692

OR

$$\boxed{\bar{X}_{AC_A} = .00479 \, S_H - .0233}$$

This is plotted in Figure J.2

C.G. SHIFT DUE TO INCREASING TAIL AREA

$\frac{W_{emp}}{S_{emp}} = 4.482$ psf from Class I Weights

$W_v = 761.9$ lbs

Thus

$W_{emp} = 4.482 S_H + 761.9$

S_H	W_{EMP}	$\bar{X}_{CG_{AFT}}$	$\bar{X}_{cg_{AFT}}$
50	986	663	.235
100	1210	672	.335
150	1434	680	.425
200	1658	689	.525

$$\boxed{\bar{X}_{cg_{AFT}} = .00192 S_H + .140}$$

This is plotted in Fig. J.2

CALCULATION OF $C_{L_{\alpha V}}$

$A = 1.4 \quad \Lambda_{LE} = 40^\circ \quad \Lambda_{c/2} = 19.9^\circ$

$$K = 1 + (1.87 - .000233\,\Lambda_{LE})A/100$$

$$K = 1.026$$

Using the Polhamus equation from Figure 3.12 of Ref. 5 again:

At M = .70 $\quad \boxed{C_{L_{\alpha V}} = 1.87 \text{ RAD}^{-1}}$

At M = 0 $\quad \boxed{C_{L_{\alpha V}} = 1.89 \text{ RAD}^{-1}}$

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CALCULATION OF Cn_{β_B} (Using method of Ref. 6)

$$Cn_{\beta_B} = -57.3\, K_N K_{R_\ell} \frac{S_{B_S}}{S} \frac{\ell_B}{b}$$

ℓ_B = 94.6 ft

S = 592 ft^2

b = 84.3 ft

S_{B_S} = 624 ft^2

X_m = 47.7 ft

h_1 = 98 in

h_2 = 89 in

h = 98 in

w = 98 in

K_N = .001

$R_\ell = \frac{\rho V \ell_f}{\mu}$ V = 696 fps

$R_\ell = 189 \times 10^6$

$K_{R_\ell} = 2.08$

$$\boxed{Cn_{\beta_B} = -.141 \text{ RAD}^{-1}}$$

Calculation of Cn_{β_V}

$$Cn_{\beta_V} = C_{L_{\alpha V}} \frac{S_V}{S} \frac{X_V}{b} = 1.87(S_V/592)(37.17/84.3)$$

$$\boxed{Cn_{\beta_V} = .00139\, S_V}$$

Calculation of C_{n_β}

$$C_{n_\beta} = C_{n_{\beta_B}} + C_{n_{\beta_V}}$$

Thus, $\boxed{C_{n_\beta} = -.141 + .00139 S_v}$

This is plotted in Fig. J.3 .

For a value of $C_{n_\beta} = .001$ DEG^{-1} an $S_v = 143$ ft^2 is required.

Estimating $C_{n_{\delta_R}}$* for Engine - Out Calculation

$$C_{y_{\delta_R}} = C_{L_{\alpha_V}} \frac{(\alpha_\delta)_{C_L}}{(\alpha_\delta)_{C_\ell}} (\alpha_\delta)_{C_\ell} K' K_b \frac{S_V}{S}$$

$C_{L_{\alpha_V}} = 1.89 \text{ RAD}^{-1}$ at $M=0$

$(\alpha_\delta)_{C_\ell} = -.70$ $\qquad \frac{(\alpha_\delta)_{C_L}}{(\alpha_\delta)_{C_\ell}} = 1.11$

$K_b = 1.0$ $\qquad K' = .65$

$C_{y_{\delta_R}} = -.00152\, S_V \quad \text{RAD}^{-1}$

$$C_{n_{\delta_R}} = -C_{y_{\delta_R}} \frac{X_V}{b}$$

$$= -.00152\, S_V \left(\frac{37.17}{84.3}\right)$$

$$\boxed{C_{n_{\delta_R}} = 6.689 \times 10^{-4}\, S_V}$$

* From Method in Ref. 6

ENGINE - OUT CALCULATION

From method in Ref. 2, Section 11.3

$y_t = 5.49$ (from 3-view)

$P_{TO_{REQ}} = 6000$ (one engine)

From Fig. 3.8 of Roskam's Design Book (Part I),

$T_{TO_e} = 10{,}962$ lbs

$N_{t_{crit}} = T_{TO}\, y_{t_{eff}} = 10962(5.49)$

$N_{t_{crit}} = 60184$ ft-lbs

$N_D = .40\, N_{t_{crit}} = 24074$ ft-lbs

$V_{mc} = 1.2\, V_{s_L}$

$V_{s_L} = 141$ fps

$V_{MC} = 169.4$ fps

$\bar{q}_{MC} = 33.9$ psf

$\delta_R = (N_{t_{crit}} + N_D) / \bar{q}_{MC} S b C_{n_{\delta_R}}$

$$\boxed{\delta_R = 74.17 / S_V}$$

For a maximum rudder deflection of $\delta_R = 25° = .4363$ RAD, the required vertical tail area is:

$S_V = 170$ ft^2

REVISED VERTICAL TAIL TO HOLD ENGINE-OUT

$S_v = 170 \text{ ft}^2$

$b_v = 15.4 \text{ ft}$

$A_v = 1.4$

$c_t = 7.35 \text{ ft}$

$c_r = 14.7 \text{ ft}$

$\lambda_v = .50$

$\frac{c_r}{c_v} = .34$

$\bar{c}_v = 11.4 \text{ ft}$

$\Lambda_{LE_v} = 40°$

$y_t = 5.49$

$l_v = 37.2 \text{ ft}$

J.8 CALCULATION OF CLASS I DRAG POLARS

In this section the airplane wetted area is determined. From this, the skin friction drag is approximated. Class I drag polars are constructed and compared with the preliminary drag polars from performance sizing. Table J.9 contains a wetted area breakdown.

List of components that contribute to wetted area:

- Wing
- Vertical Tail
- Horizontal Tail
- Nacelles and Pylons
- Fuselage

1) Wing

$$S_{wet} = 2 S_{exp\,plf}\left[1 + .25(t/c)_r(1+\tau\lambda)/(1+\lambda)\right]$$

$$S_{exp} = 592 - 80.5 = 511.5 \text{ ft}^2$$

$(t/c)_r = .13$ $\quad (t/c)_t = .10$ $\quad \lambda = .40$

$\tau = 1.3$

$$S_{wet} = 2\left[511.5\left\{1 + .25(.13)(1+1.3\times.4)\right\}/(1+.40)\right]$$

$$\boxed{S_{wet} = 1059 \text{ ft}^2}$$

2) Vertical Tail

$$S_{exp} = 170 \text{ ft}^2$$

$(t/c)_r = .13$ $\quad (t/c)_t = .12$ $\quad \lambda = .5$

$\tau = 1.08$

$$S_{wet} = 2(170)\left[1 + (.25)(.13)(1+.5\times1.08)/1.5\right]$$

$$\boxed{S_{wet} = 351}$$

3) Horizontal Tail

$S_{exp} = 102 \text{ ft}^2$

$\lambda = .5 \qquad (t/c)_r = .12 \qquad (t/c)_t = .10 \qquad \tau = 1.2$

$S_{wet} = 2(102)\left[1 + .25(.13)(1+1.2\times.5)/1.5\right]$

$\boxed{S_{wet} = 207 \text{ ft}^2}$

4) Nacelles

$S_{wet} = \pi \, l_{eng} \, D_{eng}$

$= \pi(108.5)(38)/144 = 90 \text{ ft}^2$

2 ENGINES : $\boxed{S_{WET} = 180 \text{ ft}^2}$

5) Pylons

$\lambda = 1 \qquad (t/c)_r = .12 \qquad \tau = 1$

$S_{wet} = 2(5)(6)[1.03] = 62 \text{ ft}^2$

2 PYLONS : $\boxed{S_{WET} = 124 \text{ ft}^2}$

6) Fuselage

$D_f = 8.05 \text{ ft} \qquad l_f = 94.08 \text{ ft} \qquad \lambda_f = 94.08/8.05 = 11.69$

$S_{wet} = \pi D_f \, l_f \, (1-2/\lambda_f)^{2/3}(1+1/\lambda_f^2)$

$= \pi(8.05)(94.08)(1-2/11.69)^{2/3}(1+1/11.69^2)$

$\boxed{S_{wet} = 2115 \text{ ft}^2}$

TABLE J.9 WETTED AREAS OF 50PAX COMPONENTS

COMPONENT	WETTED AREA
WING	1059
V-TAIL	351
H-TAIL	207
NACELLES	180
PYLONS	124
FUSELAGE	2115
TOTAL	4036

From Fig. 3.21b) of Roskam's Design Book Part I,

for $C_f = .0030$,

$$f = 12.1 \ ft^2$$

$$C_{D_0} = f/S = 12.1/592$$

$$\boxed{C_{D_0} = .0204} \quad M=0$$

Adding .0002 to C_{D_0} for compressibility effects:

$$\boxed{C_{D_0} = .0206} \quad M=.70$$

Take-off C_{D_0} increment, $\Delta C_{D_0} = .015$

Landing C_{D_0} increment

.015 for gear

.075 for flaps

DRAG POLARS

Take-off : $C_{D_0} = .0204 + .015 = .0354$

$e = .80 \qquad A = 12$

$$C_D = C_{D_0} + \frac{C_L^2}{\pi A e}$$

$$\boxed{C_D = .0354 + .0332 C_L^2}$$

CRUISE : $C_{D_0} = .0206$

$e = .85 \qquad A = 12$

$$\boxed{C_D = .0206 + .0312 C_L^2}$$

LANDING : $C_{D_0} = .0204 + .090 = .110$

$e = .8 \qquad A = 12$

$$\boxed{C_D = .110 + .0332 C_L^2}$$

At cruise $C_L = .3$

By taking a 10% reduction in C_{D_0} by the use of NLF :

$$C_D = .0185 + .0312 C_L^2$$

$$(L/D)_{CR} = 14.1$$

The preliminary assumption was $(L/D)_{CR} = 16$. Thus $\Delta(L/D) = -1.9$

From initial weight sizing,

$$\frac{\partial W_{TO}}{\partial (L/D)} = -994 \text{ lbs}$$

Thus $\Delta W_{TO} = 994(1.9) = 1889$ lbs

$$W_{TO_{New}} = W_{TO} + \Delta W_{TO}$$

$$= 42057 + 1889$$

$$W_{TO_{NEW}} = 43,946 \text{ lbs}$$

This represents a 4.5% change in take-off weight. This magnitude of change does not warrant resizing.

APPENDIX K

ENGINEERING CALCULATIONS FOR THE 75 PASSENGER COMMUTER

TABLE OF CONTENTS

K.1 INTRODUCTION

THE PURPOSE OF THIS APPENDIX IS TO PRESENT THE PRELIMINARY SIZING AND CLASS I DESIGN CALCULATIONS. METHODS USED WERE TAKEN FROM REFERENCES 1 AND 2.

REFERENCES 5 AND 6 ARE USED FOR STABILITY AND CONTROL DESIGN CALCULATIONS

SECTION K.2 CONTAINS PRELIMINARY SIZING CALCULATIONS. THESE RESULTS ARE FROM XEWTOG, A COMPUTER PROGRAM AVAILABLE AT THE UNIVERSITY OF KANSAS.

SECTION K.3 CONTAINS PRELIMINARY PERFORMANCE RESULTS FROM XPRFRM.

SECTION K.4 CONTAINS CLASS I FLAP SIZING CALCULATIONS.

SECTION K.5 CONTAINS CLASS I EMPENNAGE SIZING ($\bar{V}$-METHOD).

SECTION K.6 CONTAINS LANDING GEAR DESIGN CRITERIA.

SECTION K.7 CONTAINS STABILITY AND CONTROL CALCULATIONS.

SECTION K.8 CONTAINS THE WETTED AREA CALCULATIONS AND THE CLASS I DRAG POLARS.

K.2 INITIAL WEIGHT SIZING

USING XEWTOG, A WEIGHT SIZING PROGRAM WHICH FOLLOWS THE METHOD IN CH. 2 OF REFERENCE 1, THE FOLLOWING WEIGHTS AND TAKE-OFF WEIGHT SENSITIVITIES ARE FOR THE 75 PASSENGER AIRPLANE. SEE TABLE K. 1

THE DESIGN ASSUMPTIONS USED IN THE WEIGHT SIZING ARE :

$$(L/D)_{CR} = 16$$

$$C_P = 0.4 \text{ lbs/HP/hr}$$

$$\eta_P = 0.85$$

$$V_{CR} = 442 \text{ Knots}$$

TABLE K.1a : INITIAL WEIGHT SIZING RESULTS

```
WEIGHT ESTIMATION FOR A REGIONAL,
PROPELLER DRIVEN AIRCRAFT
PASSENGER WEIGHT IS     15775.
CARGO WEIGHT IS             0.
CREW WEIGHT IS            520.
PHASE W/W  CJ OR CP    NP      L/D      ALTCR       RC    MCR OR V    E OR R     PLDROP
  1    0.990   0.00   0.00     0.00       0.        0.0      0.00        0.00         0.
  2    0.995   0.00   0.00     0.00       0.        0.0      0.00        0.00         0.
  3    0.995   0.00   0.00     0.00       0.        0.0      0.00        0.00         0.
  4    0.985   0.00   0.00     0.00       0.        0.0      0.00        0.00         0.
  5    0.873   0.40   0.85    14.00       0.        0.0      0.00     1500.00         0.
  7    0.985   0.00   0.00     0.00       0.        0.0      0.00        0.00         0.
  8    0.995   0.00   0.00     0.00       0.        0.0      0.00        0.00         0.
REGRESSION COEFFICIENTS ARE A=0.3959 AND B=0.9647
THE MISSION FUEL FRACTION WITHOUT RESERVES IS:0.325
THE GROSS TAKE OFF WEIGHT IS      82491. POUNDS.
THE EMPTY WEIGHT IS        48175. POUNDS.
THE WEIGHT OF FUEL IS      17898. POUNDS.
```

TABLE K.16 : SENSITIVITY RESULTS

```
**EMPTY WEIGHT REDUCTION DUE TO COMPOSITES: 5.0 PER CENT
** SENSITIVITY ANALYSIS BEGINS HERE **
GROWTH FACTOR DUE TO PAYLOAD WEIGHT IS..............          5.7
THE TAKE-OFF WEIGHT TO EMPTY WEIGHT SENSITIVITY IS            1.7


CHOICE NUMBER..        5
                    CRUISE
SFC (LB/LB/HR)                 ****
SFC (LB/HP/HR)                 0.40
PROP EFFICIENCY               0.85
L/D                           16.0
VELOCITY (KNOTS)               0.0
RANGE (NT. MILES)           1500.0
ENDURANCE (HRS)            *******
THE SENSITIVITY OF GROSS TAKE-OFF WEIGHT TO THE FOLLOWING PARAMETERS
IS NOW GIVEN AS THE PARTIAL DERIVATIVE OF THE GROSS TAKE-OFF WEIGHT
TO THE INDICATED PARAMETER.
DWTO/DCJ (LB/LB/LB/HR)          0.0
DWTO/DCP (LB/LB/HP/HR)     143159.0
DWTO/DNP (POUNDS)          -67382.1
DWTO/D(L/D) (POUNDS)        -3579.7
DWTO/DV (LB/KNOT)               0.0
DWTO/DR (LB/NT MILE)           30.2
DWTO/DE (LB/HP)                 0.0
```

K.3 INITIAL PERFORMANCE SIZING

THE RESULTS FROM XPRFRM, A PERFORMANCE SIZING PROGRAM, ARE PRESENTED IN THIS SECTION, THE METHODS USED ARE IN CH. 3 OF REFERENCE 2. SEE TABLES K.2 THROUGH K.6.

TABLE K.2

```
*****************************************************
****************  TAKE-OFF SIZING  ****************
*****************************************************

          "FAR 25 CERTIFICATION CATEGORY"

                        REGIONAL TURBO-PROP

                    PROPELLER DRIVEN

------------------  INPUT DATA  ------------------

ALTITUDE                                  0.0(FEET)
FAR 23 TAKE-OFF DISTANCE <STO>            0.0 (FEET)
MINIMUM WING LOADING                   20.00(LB/FT**2)
MAXIMUM WING LOADING                  100.00(LB/FT**2)
MINIMUM TAKE-OFF LIFT COEFFICIENT 1.60
MAXIMUM TAKE-OFF LIFT COEFFICIENT 2.40

-------------------  OUTPUT DATA  ----------------

          TABLE OF POWER LOADINGS

 W/S            CLMAX-TO

0.00     1.60   1.80   2.00   2.20   2.40

 20.0    21.5   24.2   26.9   29.6   32.3
 40.0    10.8   12.1   13.5   14.8   16.2
 60.0     7.2    8.1    9.0    9.9   10.8
 80.0     5.4    6.1    6.7    7.4    8.1
100.0     4.3    4.8    5.4    5.9    6.5
                        REGIONAL TURBO-PROP

                     FAR 25 CERTIFICATION CATEGORY

GROSS TAKE-OFF WEIGHT  (WTO)      82491.0(LBS)
LANDING TO TAKE-OFF WEIGHT RATIO      1.000
ALTITUDE                              0.0(FEET)
DENSITY                            .0023769(SLUG/FT**3)
LANDING APPROACH SPEED  (VA)        108.0(KTS)
LANDING FIELD LENGTH  (SFL)        3500.0(FEET)

        (W/S)TO= 23.40CLMAX(LAND)

       MAXIMUM TAKE-OFF WING LOADINGS
     TO MEET LANDING DISTANCE REQUIREMENT
    ======================================

     CLMAX        MAXIMUM WING LOADING




     (LAND)          (TAKE-OFF)
                     (LB/FT**2)
     --------     -----------------

      2.20             51.49
      2.40             56.17
      2.60             60.85
      2.80             65.53
      3.00             70.21
```

TABLE K.3

```
 ********************************************
 ***********   SIZING TO STALL SPEED   *********
 ***********                           *********
 ***********   MAX T-O WING LOADING    *********
 ********************************************


    TAKE-OFF WEIGHT                 82491.0  LBS
    STALL SPEED (FLAPS-UP)            120.00  KTS
    STALL SPEED (FLAPS-DOWN)          100.00  KTS
    CLMAX (FLAPS-UP)                    1.50
    CLMAX (FLAPS-DOWN)                  3.00

    WING LOADING (FLAPS-UP)            73.22  LB/FT**2
    WING LOADING (FLAPS-DOWN)         101.70  LB/FT**2


    MAX TAKE-OFF WING LOADING          73.22  LB/FT**2

 ***********************************************
 **************   DRAG POLAR EQUATIONS   ***********
 ***********************************************


          ******   INPUT DATA   ******

    MAXIMUM TAKE-OFF WEIGHT (CLEAN)     82491.0  (LBS)
    WING AREA                             500.00  (FT**2)
    ASPECT RATIO                           12.00
    SKIN FRICTION COEFFICIENT            0.00250
    AIRPLANE WETTED AREA                 8173.(FT**2)

  DRAG INCREMENT DUE TO TAKE-OFF FLAPS   .0150
  DRAG INCREMENT DUE TO LANDING FLAPS    .0300
  DRAG INCREMENT DUE TO LANDING GEAR     .0200

  OSWALDS EFFICIENCY FACTOR (CLEAN)      .850
  OSWALDS EFFICIENCY FACTOR (TAKE-OFF)   .800
  OSWALDS EFFICIENCY FACTOR (LANDING)    .800

          ******   CALCULATED DATA   ******

  THE COMPLETE SET OF DRAG POLARS IS:

     1.  LOW-SPEED  (CLEAN):
     CD = .0490 + .0312CL**2        L/Dmax =12.78

     2.  TAKE-OFF  (LANDING GEAR UP):
     CD = .0640 + .0332CL**2        L/Dmax =10.85

     3.  TAKE-OFF  (LANDING GEAR DOWN):
     CD = .0840 + .0332CL**2        L/Dmax = 9.47

     4.  LANDING  (LANDING GEAR UP):
     CD = .0790 + .0332CL**2        L/Dmax = 9.77



    5.  LANDING  (LANDING GEAR DOWN):
    CD = .0990 + .0332CL**2        L/Dmax = 8.73

 ************************************************
```

TABLE K.4

```
FAR 25.111 (CEI)  "INITIAL CLIMB SEGMENT"
===========================================

FAR 25.111 CLIMB GRADIENT (INITIAL SEGMENT)    1.2000
                  TABLE OF POWER LOADINGS REQUIRED
                  --------------------------------

WING LOADING =  20.00      40.00      60.00      80.00     100.00
   (LB/FT**2)

ASPECT
RATIO




 10.00              77.75      54.98      44.89      38.88      34.77
 11.00              81.87      57.89      47.27      40.94      36.61
 12.00              85.66      60.57      49.45      42.83      38.31
 13.00              89.14      63.03      51.47      44.87      39.87
 14.00              92.37      65.31      53.33      46.16      41.31
FAR 25.121 (CEI)  "TRANSITION SEGMENT CLIMB"
============================================

FAR 25.121 CLIMB GRADIENT (TRANSITION)          0.0001
                  TABLE OF POWER LOADINGS REQUIRED
                  --------------------------------

WING LOADING =  20.00      40.00      60.00      80.00     100.00
   (LB/FT**2)

ASPECT
RATIO

 10.00              81.59      57.70      47.11      40.80      36.49
 11.00              86.63      61.26      50.02      43.32      38.74
 12.00              91.33      64.58      52.73      45.66      40.84
 13.00              95.72      67.68      55.26      47.86      42.81
 14.00              99.83      70.59      57.64      49.92      44.65
FAR 25.121 (CEI)  "SECOND SEGMENT CLIMB"
========================================

FAR 25.121 CLIMB GRADIENT (SECOND SEGMENT)     2.4000
                  TABLE OF POWER LOADINGS REQUIRED
                  --------------------------------

WING LOADING =  20.00      40.00      60.00      80.00     100.00
   (LB/FT**2)

ASPECT
RATIO

 10.00              70.67      49.97      40.80      35.33      31.60
 11.00              74.06      52.37      42.76      37.03      33.12
 12.00              77.14      54.55      44.54      38.57      34.50
 13.00              79.96      56.54      46.16      39.98      35.76
 14.00              82.54      58.36      47.65      41.27      36.91
```

TABLE K.5

```
FAR 25.121 (OEI)  "EN-ROUTE CLIMB SEGMENT"
==========================================

FAR 25.121 CLIMB GRADIENT (EN-ROUTE)             1.2000

                    TABLE OF POWER LOADINGS REQUIRED
                    --------------------------------

WING LOADING =  20.00       40.00       60.00       80.00      100.00
   (LB/FT**2)

ASPECT
RATIO



 10.00                   67.78       47.93       39.13       33.89       30.31
 11.00                   69.98       49.48       40.40       34.99       31.29
 12.00                   71.92       50.85       41.52       35.96       32.16
 13.00                   73.64       52.07       42.52       36.82       32.94
 14.00                   75.19       53.17       43.41       37.60       33.63
FAR 25.119 (AEO)  "LANDING CLIMB SEGMENT"
=========================================

FAR 25.119 CLIMB GRADIENT (LANDING)              3.2000

                    TABLE OF POWER LOADINGS REQUIRED
                    --------------------------------

WING LOADING =  20.00       40.00       60.00       80.00      100.00
   (LB/FT**2)

ASPECT
RATIO

 10.00                   30.32       21.44       17.51       15.16       13.56
 11.00                   31.60       22.35       18.25       15.80       14.13
 12.00                   32.76       23.16       18.91       16.38       14.65
 13.00                   33.80       23.90       19.52       16.90       15.12
 14.00                   34.75       24.57       20.06       17.38       15.54
```

TABLE K.6

```
FAR 25.121 (OEI)  "GO-AROUND OR BALKED LANDING"
================================================

FAR 25.121 CLIMB GRADIENT (GO-AROUND)             2.1000
                     TABLE OF POWER LOADINGS REQUIRED
                     --------------------------------

WING LOADING =  20.00       40.00       60.00       80.00     100.00
   (LB/FT**2)

ASPECT
RATIO

 10.00             52.95       37.44       30.57       26.47       23.68
 11.00             54.53       38.56       31.49       27.27       24.39
 12.00             55.93       39.55       32.29       27.97       25.01
 13.00             57.17       40.43       33.01       28.59       25.57
 14.00             58.28       41.21       33.65       29.14       26.06

ALTITUDE                                 30000.0 (FEET)
DENSITY                                 .0008893 (SLUG/FT**3)
ASPECT RATIO                               12.00
OSWALD'S EFFICIENCY FACTOR (e)              0.85
PROPELLER EFFICIENCY                        0.90
ZERO LIFT DRAG COEFFICIENT               0.04903
MAXIMUM TAKE-OFF WEIGHT                  62491.0 (LBS)
SPECIFIED MANEUVERING WEIGHT             50000.0 (LBS)

VELOCITY                                   250.0 (KTS)
MACH NUMBER                                 0.42
MAXIMUM LOAD FACTOR                         1.00 (g's)




MINIMUM WING LOADING                       20.00 (LB/FT**2)
MAXIMUM WING LOADING                      100.00 (LB/FT**2)

          (W/P)=(W/S)/   0.30 + 2978.35(W/S)

                  POWER LOADINGS NECESSARY
             TO MEET THE CRUISE SPEED REQUIREMENTS
 ===============================================================

   (W/S)      (W/S)       (W/P)         (W/P)        (W/P)
  ACTUAL    TAKEOFF      ACTUAL       TAKEOFF      TAKEOFF
                                     IN FLIGHT      STATIC
   (psf)      (psf)      (lb/hp)      (lb/hp)      (lb/hp)
 ---------------------------------------------------------------

    17.00      20.00        2.10         2.47         1.24
    34.00      40.00        4.20         4.94         2.47
    51.00      60.00        6.30         7.42         3.71
    68.00      80.00        8.41         9.89         4.94
    85.00     100.00       10.51        12.36         6.18
```

K.4 FLAP SIZING

USING A METHOD IN CH. 7 OF REF 2, IT WAS DETERMINED THAT THE FOLLOWING FLAP GEOMETRY WOULD SUPPLY THE INCREMENTAL LIFT NECESSARY FOR TAKE-OFF AND LANDING. SEE TABLE K.7. THE DESIGN CALCULATIONS FOLLOW.

TABLE K.7 75 PAX FLAP GEOMETRY

TRAILING EDGE FOWLER FLAPS

$c_f/c = 0.25$

$S_{wf}/S \approx 0.80$

$b_f/b \approx 0.80$

$\delta_f = 30°$

FLAP SIZING : 75 PAX

$$\tan \Lambda_{.25\bar{c}} = \tan \Lambda_{LE} - 4/12 \left[(.25)(.429)\right]$$

$$\tan \Lambda_{.25\bar{c}} = .268 - .036$$

$$\Lambda_{.25\bar{c}} = 13^\circ$$

ASSUME FROM MATCHING GRAPH :

$$C_{L_{MAX_{CLEAN}}} = 1.40$$

$$C_{L_{MAX_W}} = 1.05 \left(C_{L_{MAX_{CLEAN}}}\right) = 1.05(1.40)$$

$$C_{L_{MAX_W}} = 1.47$$

$$C_{L_{MAX_{W_{UNSWEPT}}}} = 1.47 / \cos 13^\circ$$

$$C_{L_{MAX_{W_{UNSWEPT}}}} = 1.51$$

$$C_{L_{MAX_W}} = 0.95(1.5 + 1.5)/2$$

± 10% OK

$$C_{L_{MAX_W}} = 1.43$$

$$\Delta C_{L_{MAX_L}} = 1.05 \left(C_{L_{MAX_L}} - C_{L_{MAX_W}}\right)$$

$$\Delta C_{L_{MAX_L}} = 1.05(3.00 - 1.43)$$

$$\Delta C_{L_{MAX_L}} = 1.65$$

$$\Delta C_{l_{MAX}} = \Delta C_{L_{MAX}} (S/S_{wf}) K_\Lambda$$

$K_\Lambda = (1 - 0.08 \cos^2 13°) \cos^{3/4} 13°$

$K_\Lambda = 0.906$

ASSUME : $c_f/c = 0.25$

$\delta_f = 30°$

$K = 0.96$

$\Delta C_l = (1/0.96) \Delta C_{l_{MAX}}$

S_{wf}/S	S/S_{wf}	$\Delta C_{l_{MAX}}$	ΔC_l
0.4	2.50	3.74	3.90
0.5	2.00	2.99	3.11
0.6	1.67	2.50	2.60
0.7	1.43	2.14	2.23
0.8	1.25	1.87	1.95
0.9	1.11	1.66	1.73

For Fowler Flaps :

ASSUME : $C_{l_\alpha} = 2\pi$

$C_{l_{\alpha_f}} = 2\pi(1 + 0.25) = 7.85$

$\Delta C_l = C_{l_{\alpha f}} \alpha_{\delta_f} \delta_f = (7.85)(0.47)(0.52)$

$\Delta C_l = 1.93$

$\therefore \; S_{wf}/S \cong 0.8$

K.5 $\bar{V}$-METHOD FOR EMPENNAGE SIZING

THE 75 PAX COMMUTER EMPENNAGE IS OF THE CONVENTIONAL T-TAIL TYPE. TABLE K.8 CONTAINS THE GEOMETRY OF THE EMPENNAGE. FROM TABLES 8.6 a AND 8.6 b, AVERAGE VALUES OF $\bar{V}_h$ AND $\bar{V}_v$ WERE OBTAINED.

EMPENNAGE SIZING 75 PAX:

THE FOLLOWING VALUES WERE DETERMINED BY COMPARISON WITH MD-80, B727-200, BAe 146-200 and the FOKKER F28.

$l_f = 108.42$ ft.

$d_f = 8.05$ ft.

$X_v = 45.5$ ft.

$X_h = 55.3$ ft.

$\overline{V}_h = 1.08$

$\overline{V}_v = 0.083$

$S_h = 241.8\ ft^2$

$S_v = 255.5\ ft^2$

$S_e = 87.0\ ft^2$

$S_r = 86.9\ ft^2$

ELEVATOR CHORD ROOT/TIP (FRACTION OF C_h)

0.39/0.45

RUDDER CHORD ROOT/TIP (FRACTION OF C_v)

0.35/0.32

HORIZONTAL TAIL :

TABLE K.8

$\Gamma = 0°$

i_h = VARIABLE

$AR = 5.3$

$\Lambda_{.25\bar{c}} = 22°$

$\lambda_h = 0.35$

$C_r = 11.01$ ft

$C_t = 3.85$ ft

$b = 32.53$ ft

VERTICAL TAIL :

$\Gamma = 90°$

$i_v = 0$

$AR = 1.4$

$\Lambda_{.25\bar{c}} = 42°$

$\lambda_v = 0.60$

$C_r = 19.64$ ft

$C_t = 11.79$ ft.

$b = 16.26$ ft.

K.6 CLASS I LANDING GEAR DESIGN

FROM CH. 9 IN REFERENCE 2 IT WAS DECIDED TO CHOOSE A 30" DIAMETER TIRE 9" WIDE. THIS TIRE CAN CARRY 20,000 LBS.

FROM WEIGHT AND BALANCE CALCULATIONS LONGITUDINAL GEAR PLACEMENT CRITERION WERE MET. THERE IS A 15° BETWEEN GROUND CONTACT POINT AND AFT C.G.

FIGURE K.1 SHOWS THAT THE LATERAL TIP-OVER CRITERION IS MET FOR A 206" WHEEL BASE.

1.) TIP-OVER CRITERIA:

FOR TRICYCLE GEARS: THE MAIN GEAR MUST BEHIND THE AFT C.G. LOCATION. THE USUALLY RELATIONSHIP BETWEEN THE AFT C.G. LOCATION AND THE MAIN LANDING GEAR IS 15°

IN THE FIGURE, IT IS CLEAR THAT THE MAIN GEAR IS LOCATED BEHIND THE AFT C.G. LOCATION. ALSO, IT IS CLEAR THAT THE AFT C.G. LOCATION IS LOCATED FORWARD OF THE 15° ANGLE WITH REFERENCE TO THE MAIN GEAR. THE LOCATION OF THE GEAR SATISFIES THE LONGITUDINAL TIP-OVER CRITERION

2) LATERAL TIP-OVER CRITERIA

THE LATERAL TIP-OVER IS DICTATED BY THE ANGLE φ.

CLASS I METHOD FOR LANDING GEAR SIZING AND DISPOSITION

STEP 9.1 DECIDE WHICH LANDING GEAR SYSTEM TO USE:

RETRACTABLE

STEP 9.2 DECIDE ON THE OVERALL LANDING GEAR CONFIGURATION:

CONVENTIONAL (TRICYCLE)

STEP 9.3 PROCEED TO CHAPTER 10 AND PREPARE A ROUGH WEIGHT AND BALANCE STATEMENT FOR AN ASSUMED DISPOSITION OF THE LANDING GEAR

SEE FIGURE

STEP 9.4 DECIDE ON A PRELIMINARY LANDING GEAR STRUT DISPOSITION AND SKETCH THE PROPOSED STRUT DISPOSITION IN THE GENERAL ARRANGEMENT DRAWING OF STEP 10.2 (CHAPTER 10).

SEE FIGURE

FOR $\psi = 45^\circ$, $B = 150''$

$\psi = 50^\circ$, $B = 124''$

$\psi = 55^\circ$, $B = 103''$

NOTE

$\psi \leq 55^\circ$

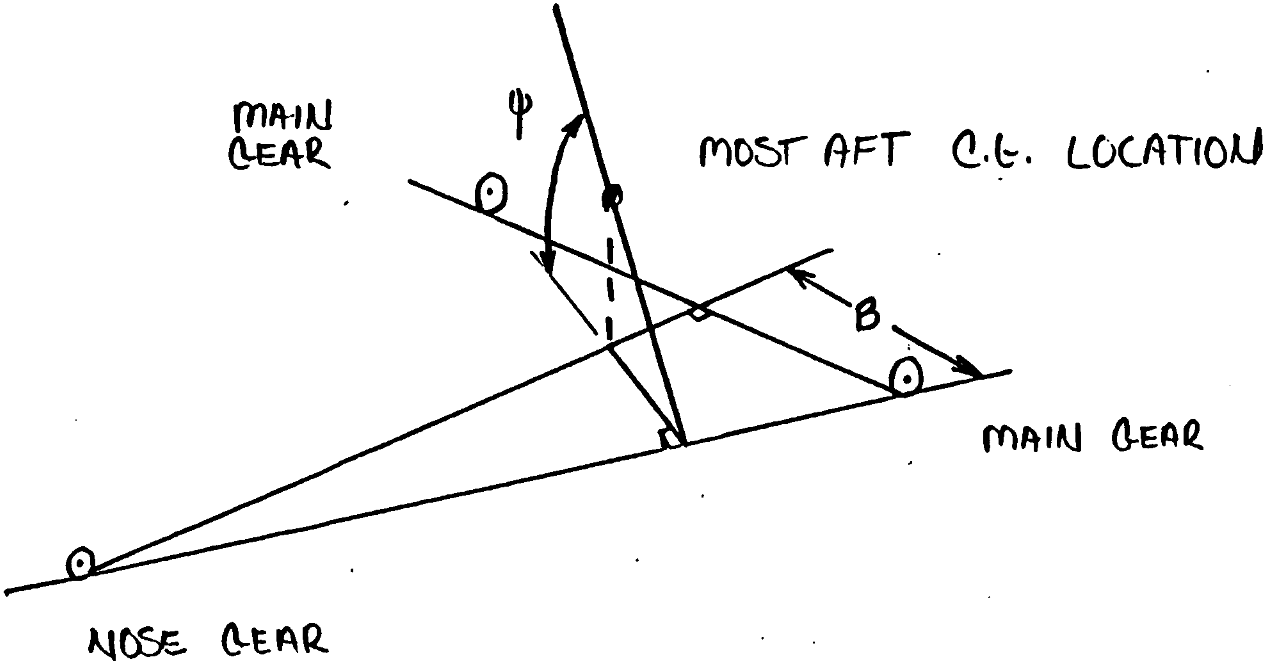

FIGURE K.1

K.7 STABILITY AND CONTROL CALCULATIONS

CALCULATION OF REQUIRED STABILITY AND CONTROL DERIVATIVES ARE PRESENTED IN THIS SECTION.

Body Segment	W_f	X_i	$W_f^2 (X_i)$	ΔX_i	$d\varepsilon/d\alpha$
1 (cockpit)	70	595	34.03	8.33	1.03
2	96	495	64.0	8.33	1.03
3	96	395	64.0	8.33	1.06
4	96	295	64.0	8.33	1.13
5	96	195	64.0	8.33	1.20
6	96	72.5	64.0	12.08	1.89
WING					
7	96	91	64.0	15.17	.144
8	94	232	61.36	8.33	.367
9	78	332	42.25	8.33	.526
10	50	439.5	17.36	9.58	.696
E_1^{Engine}	30	434	6.25	10.92	.687
E_2^{Engine}	30	434	6.25	10.92	.687

$\Lambda_{c/2} = 11.12°$

$\beta = 0.71414$

$\kappa = .682$

$K = 1.05439$

$C_{L_{\alpha_W}} = 4.709 \text{ rad}^{-1} = .0822$ (REF. 5)

$$\frac{d\bar{\varepsilon}}{d\alpha}_{C_{L\alpha}} = \frac{d\bar{\varepsilon}}{d\alpha}_{C_{L\alpha}=.08} \times 1.0274$$

$c_f = 172.0''$ $\quad x = 1.00 \quad d\varepsilon/d\alpha = .145$

$\lambda = .33 \quad d\varepsilon/d\alpha = .185$

$m = .33$

$r = .93$ $\quad d\varepsilon/d\alpha = .180 \times .975$

$l_H = 518''$ $\quad 1 - d\varepsilon/d\alpha = .82$

$dM/d\alpha = \bar{q}\ 132.37$

$\Delta X_{AC_B} = -.13$

$AR_H = 4.368$

$\Lambda_{.25\bar{c}} = 22°$

$\Lambda_{LE} = 27.2°$

$\Lambda_{c/2} = 16.4°$

$K = 1.06464$

$\beta = .71414$

$\kappa = .682$

$C_{L_{\alpha_H}} = 3.514 \ rad^{-1}$

$X_{oc_H} = 614.0''$

$\overline{X}_{ac_H} = 4.87$

$$X_{ac_A} = \frac{.12 + \left(\frac{3.514}{4.709}\right) \frac{S_H}{S} (4.87)(.82)}{1 + \frac{3.514}{4.709} \frac{S_H}{S} (.82)}$$

$$X_{ac_A} = .12 + .00253\, S_H \Big/ 1 + .000519\, S_H$$

$\overline{X}_{AC_A}$	S_H
.650	241.8
.567	200
.463	150
.355	100
.240	50

From Figure K.2 with SM = −0.05

$$\boxed{S_H = 134 \text{ ft}^2}$$

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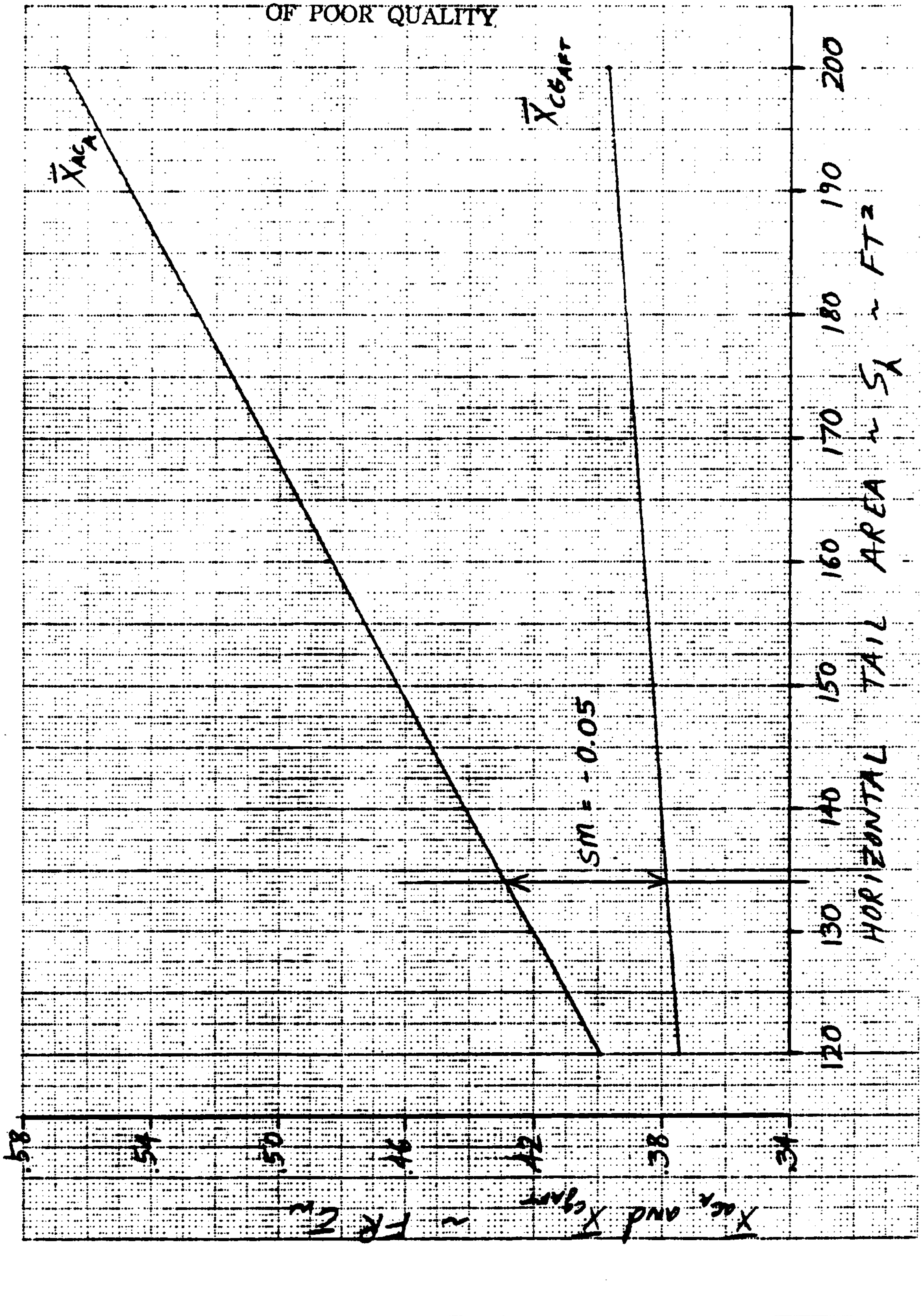


CALC	G.Dragush	12/10	REVISED	DATE	FIGURE K.2 LONGITUDINAL X-PLOT FOR THE 75 PASSENGER COMMUTER	Figure 5
CHECK						AE: 790
APPD						
APPD						
					ROSKAM AVIATION AND ENGINEERING CORPORATION	PAGE K.28

$$C_{n_{\beta_B}} = -57.3 \cdot K_N K_{R_l} \frac{S_{B_S}}{S} \frac{l_B}{b}$$

$X_m = 746$ in

$l_m = 1,301$ in

$\frac{X_m}{l_m} = 0.573$

$S_{B_S} = 734.5 \text{ ft}^2$

$K_N = 0.000875$

$K_{R_l} = 2.1$

$\mu = 3.106 \times 10^{-7}$ lb sec/ft²

$\therefore C_{n_{\beta_B}} = -0.0599$

Determine $C_{L_{\alpha_V}}$ from Polhamus:

$AR_V = 1.04$

$\Lambda_{LE} = 49°$

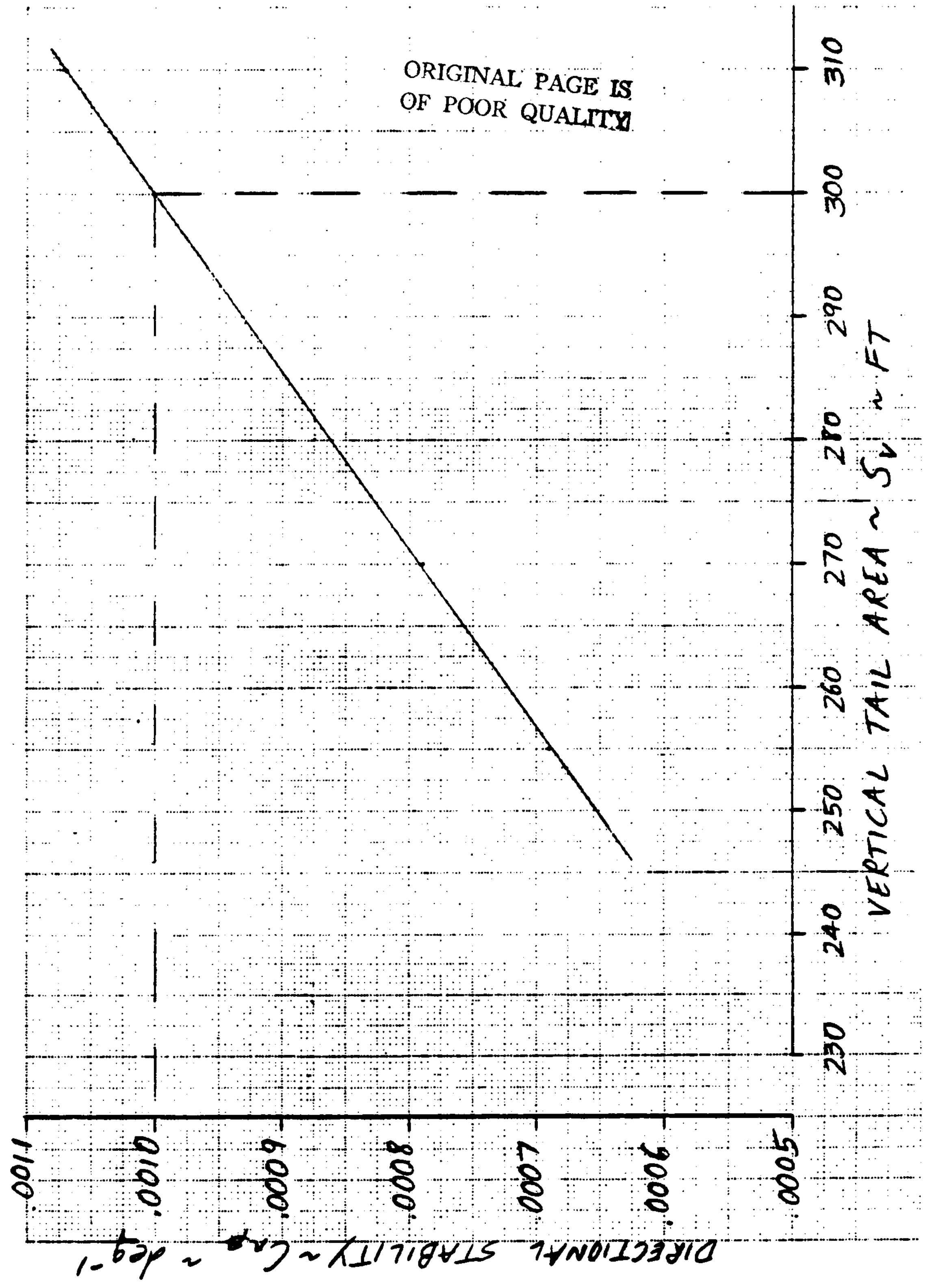
$\Lambda_{c/2} = 33.8°$

$K_V = 1.0194$

$\therefore C_{L_{\alpha_V}} = 1.426$

$$C_{n_\beta} = C_{n_{\beta_B}} + C_{L_{\alpha_V}} \left(\frac{S_V}{S}\right)\left(\frac{X_V}{b}\right)$$

$$C_{n_\beta} = -0.0599 + 0.0003902 \, S_V$$



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APPD				

FIGURE K.3
DIRECTIONAL X-PLOT
FOR THE 75 PASSENGER COMMUTER

Figure 6

AE: 790

S_v	C_{n_β} rad^{-1}	C_{n_β} deg^{-1}
255	0.0396	0.00069
200	0.01814	0.00032
270	0.04545	0.00079
300	0.05716	0.00100 ←
310	0.06106	0.00107

From S + C handbook of Reference 6

$(\alpha\delta)_{c_l} = -0.71$

$(\alpha\delta)_{c_L} / (\alpha\delta)_{c_l} = 1.02$

For $\delta_r = 25°$ and $c_r/c = 0.35$:

$K_b = 1.0$

$K' = 0.65$

$$C_{y\delta_r} = C_{L_{\alpha_v}} \frac{(\alpha\delta)_{c_L}}{(\alpha\delta)_{c_l}} (\alpha\delta)_{c_l} K' K_b \frac{S_v}{S}$$

$$C_{n\delta_r} = -C_{y\delta_r} \frac{l_v}{b}$$

$\therefore C_{n\delta_r} = 0.000176 \, S_v$

$C_{n\delta_r}$ needed for 25 deg rudder deflection:

$25° = 0.06386$ rad

$0.06386 = 0.000176 \, S_v$

$\therefore S_v = 363$ ft^2

K.8 CALCULATION OF CLASS I DRAG POLARS

THIS SECTION COMPUTES THE AIRPLANE WETTED AREA, AND ESTIMATES SKIN FRICTION DRAG. CLASS I DRAG POLARS ARE CONSTRUCTED AND COMPARED WITH THE POLARS COMPUTED FOR THE PERFORMANCE SIZING. TABLE K.9 CONTANS A WETTED AREA BREAKDOWN.

1) WING :

$S_{exp} = 1071.0\ ft^2$

$(t/c) = 0.13$

$\tau = 1$

$\lambda = 0.4$

$S_{WET} = 2212.0\ ft^2$

2) HORIZONTAL TAIL :

$\lambda = 0.35$

$\tau = 1.0$

$S_{WEt} = 277\ ft^2$

3) VERTICAL TAIL :

$\lambda = 0.60$

$\tau = 1.0$

$S_{WET} = 750\ ft^2$

FUSELAGE :

$\lambda_f = 13.55$

$D_f = 8.0$ ft

$l_f = 108.4$ ft

$S_{WET} = 2463$ ft^2

ENGINES :

$l = 150.1''$

$D = 37.3''$

$S_{WET} = 248$ ft^2 (2 NACELLES)

PYLONS :

$\lambda = 1.0$

$(t/c) = 0.12$

$\tau = 1.0$

$S_{WET} = 124$ ft^2 (2 PYLONS)

TOTAL BY SUMMING ALL INDIVIDUAL CONTRIBUTIONS IS :

$$S_{WET} = 6{,}074 \ ft^2$$

FROM FIGURE 3.2.1

$$f = 14.88 \ ft^2$$

$$C_{D_0} = f/S = 0.0126$$

$$\Delta C_{D_{0\,TO}} = 0.015$$

$$\Delta C_{D_{0\,L}} = 0.075$$

$$\Delta C_{D_0\,gear} = 0.020$$

DRAG POLARS

TAKE-OFF "GEAR-UP"

$AR = 12 \qquad e = 0.80$

$$\cdot \ C_D = 0.0276 + 0.0332 \ C_L^2$$

TAKE-OFF "GEAR-DOWN"

$AR = 12 \qquad e = 0.80$

$$\cdot \ C_D = 0.0476 + 0.0332 \ C_L^2$$

CLEAN "LOW-SPEED"

$AR = 12 \qquad e = 0.85$

$C_D = 0.0126 + 0.0312\ C_L^2$

LANDING "GEAR-UP"

$AR = 12 \qquad e = 0.80$

$C_D = 0.0876 + 0.0332\ C_L^2$

LANDING "GEAR-DOWN"

$AR = 12 \qquad e = 0.80$

$C_D = 0.1076 + 0.0332\ C_L^2$

$C_{L_{CRUISE}} = 0.3$

$(L/D)_{CR} = 19.47$

FROM INITIAL WEIGHT SIZING AN $(L/D)_{CR} = 16.0$ WAS ASSUMED AND

$\partial W_{TO} / \partial L/D = -3579.7$ lbs.

$\Delta L/D =$ CLASS I L/D − INITIAL L/D

$\Delta L/D = 19.47 - 16.0$

$\Delta L/D = 3.47$

$\Delta W_{TO} = \partial W_{TO} / \partial L/D \ \Delta L/D = (-3579.7)(3.47)$

$\Delta W_{TO} = -12{,}422$ LBS (DECREASE)

$W_{TO_{NEW}}$ = 82,491 LBS. - 12,422 LBS

$W_{TO_{NEW}}$ = 70,069 LBS.

APPENDIX L

ENGINEERING CALCULATIONS FOR THE 100 PASSENGER COMMUTER

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TABLE OF CONTENTS

L.1 INTRODUCTION

The purpose of this appendix is to present the preliminary sizing and class I design calculations. Methods used were taken from References (1) and (2). References (5) and (6) are used for stability and control considerations.

Section L.2 contains preliminary weight sizing calculations. These results were obtained using the methods presented in Reference (1).

Section L.3 contains preliminary performance sizing calculations. These results were obtained using the methods presented in Reference (1).

Section L.4 contains class I flap sizing calculations using the methods presented in Reference (2).

Section L.5 contains class I empennage sizing using the $\overline{V}$-bar method presented in Reference (2).

Section L.6 contains landing gear desing criteria using the methods of Reference (2).

Section L.7 contains the stability and control calculations using the methods presented in Reference (2).

Section L.8 contains the class I drag polars calculated using the methods presented in Reference (2).

L.2 INITIAL WEIGHT SIZING

Using the methods presented in Reference (1) the weights and weight sensitivities for the 100 passenger airplane were calculated and are presented in Table L.1.

The design assumptions used in the weight sizing were:

(L/D)cr = 16
Cp = 0.4 lbs/hp/hr
Np = 0.85
Vcr = 442 knots

DESIGN AIRPLANE: 100 PASSENGER COMMUTER
(Type 6 AIRPLANE)

ASSUME: RANGE = 1500 nm
$L/D = 16$
$C_p = 0.4$
$\eta_p = 0.85$
$W_{tfo} = 0.005 W_{TO}$
$W_{f_{res}} = 0.25 W_{F_{USED}}$
$h = 30{,}000$ ft
$M = 0.75$
FUEL FRACTION VALUES FROM TABLE 2.1.
CLIMB: 3000 fpm

CRUISE VELOCITY:
FROM PG. 11 OF LAN & ROSKAM AERODYNAMICS,

$V_a = 994.7$ fps @ 30,000 ft.

$$M = \frac{V_{cr}}{V_a} \Rightarrow V_{CR} = 0.75(994.7)$$

$$V_{CR} = 746 \text{ fps}$$

CONVERTING TO KNOTS (REF. PG. 523),

$V_{CR} = 746$ fps $(.5921)$ = 442 kts

ESTIMATING W_{TO}, W_E, W_F:

STEP 1: MISSION PAYLOAD WEIGHT, W_{PL}, AND CREW WEIGHT, W_{CREW}.

FROM THE GIVEN SPECIFICATION:

PASSENGERS: 100 × (175 lbs + 30 lbs)
100 × 205
$W_{PL} = 20{,}500$ lbs

CREW: 4 × (175 + 30)
4 × 205
$W_{CREW} = 820$ lbs

STEP 2: GUESS VALUE OF TAKE-OFF WEIGHT, W_{TO}.

FROM JANE'S, 1974-75, SIMILAR PLANES WERE FOUND AS FOLLOWS:

AIRPLANE	W_{PL} (lbs)	W_{TO} (lbs)	$V_{CR_{MAX}}$ (kts)	RANGE (n
AÉROSPATIALE SE 210 CARAVELLE	29,100	127,870	445	1870
HAWKER-SIDDELEY 146-200	23,015	87,500	422	1730
McDONNELL-DOUGLAS DC-9, SERIES 10, MODEL 15	21,381	90,700	487	864
McDONNELL-DOUGLAS DC-9, SERIES 20	21,885	98,000	487	1213
TUPOLEV TU-134A	18,000	103,600	469	1293
DESIGN PLANE	20,500	?	442	1500

A REASONABLE ESTIMATE OF TAKE-OFF WEIGHT WOULD BE: $\underline{W_{TO} = 96{,}000 \text{ lbs.}}$

STEP 3: DETERMINE MISSION FUEL WEIGHT, W_F.

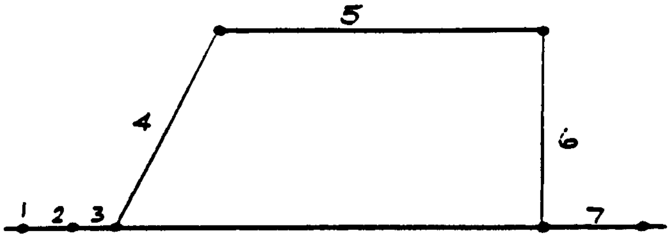
$$W_F = W_{F_{USED}} + W_{F_{RES}}$$

FROM SPEC: $W_{F_{RES}} = 0.25 W_{F_{USED}}$

$$\therefore W_F = W_{F_{USED}} + (0.25 W_{F_{USED}})$$

$$W_F = 1.25 W_{F_{USED}}$$

MISSION PROFILE:



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DEF. OF STAGES:
1) ENGINE START, WARM-UP
2) TAXI
3) TAKE-OFF
4) CLIMB AND ACCELERATE
5) CRUISE
6) DESCENT
7) LANDING, TAXI, SHUTDOWN

STEP 3: (cont.)

PHASE 1: BEGIN WEIGHT: W_{TO}
(ENGINE START) END WEIGHT: W_1

FROM TABLE 2.1 : $\frac{W_1}{W_{TO}} = \underline{0.990}$

PHASE 2: BEGIN WEIGHT: W_1
(TAXI) END WEIGHT: W_2

FROM TABLE 2.1 : $\frac{W_2}{W_1} = \underline{0.995}$

PHASE 3: BEGIN WEIGHT: W_2
(TAKE-OFF) END WEIGHT: W_3

FROM TABLE 2.1 : $\frac{W_3}{W_2} = \underline{0.995}$

PHASE 4: BEGIN WEIGHT: W_3
(CLIMB) END WEIGHT: W_4

FROM FIG 2.2 : $\frac{W_4}{W_3} = \underline{0.970}$

USING BREGUET'S ENDURANCE EQUATION, p. 13
(EQN 2.7)

$$E_{cl} = 375 \left(\frac{1}{V_{cl}}\right)\left(\frac{\eta_p}{c_p}\right)_{cl} \left(L/D\right)_{cl} \ln\left(\frac{W_3}{W_4}\right)$$

$$V_{cl} = 3000 \text{ fpm} \left(\frac{60 \text{ min}}{1 \text{ hr}}\right)\left(\frac{1 \text{ mile}}{5280 \text{ ft}}\right)$$

$$V_{cl} = 34.1 \text{ mph}$$

$$E_{cl} = \frac{30,000 \text{ ft}}{3000 \text{ fpm}} = 10 \text{ min} = \frac{1}{6} \text{ hr}$$

$$\ln\left(\frac{W_3}{W_4}\right) = \frac{E_{cl} V_{cl}}{375} \left(\frac{c_p}{\eta_p}\right)_{cl} \left(\frac{1}{L/D}\right)_{cl}$$

$$\ln\left(\frac{W_3}{W_4}\right) = \frac{\left(\frac{1}{6} \text{ hr}\right)\left(34.1 \text{ mph}\right)}{375} \left(\frac{0.4}{0.85}\right)\left(\frac{1}{16}\right) = 4.46 \times 10^{-4}$$

$\frac{W_4}{W_3} = 0.999$ ← DOESN'T SEEM CORRECT;
WILL USE ABOVE FRACTION.

STEP 3: (cont.)

PHASE 5: BEGIN WEIGHT: W_4
(CRUISE) END WEIGHT: W_5

ESTIMATE $\frac{W_5}{W_4}$ FROM BREGUET'S RANGE EQUATION, p. 15, eqn. 2.9.

$$R_{cr} = 375 \left(\eta_p / C_p\right)_{cr} \left(L/D\right)_{cr} \ln\left(\frac{W_4}{W_5}\right)$$

$$R_{cr} = 1500 \text{ nm} \left(\frac{6{,}076 \text{ ft}}{1 \text{ nm}}\right)\left(\frac{1 \text{ sm}}{5280 \text{ ft}}\right)$$

$$R_{CR} = 1726 \text{ sm}$$

$$\frac{(1726 \text{ sm})}{375} = \left(\frac{0.85}{0.4}\right)(16) \ln\left(\frac{W_4}{W_5}\right)$$

$$\ln\left(\frac{W_4}{W_5}\right) = \frac{1726}{375\,(34)} = 0.1354$$

$$\left(\frac{W_4}{W_5}\right) = 1.145 \Rightarrow \left(\frac{W_5}{W_4}\right) = 0.8734$$

$$\left(\frac{W_5}{W_4}\right) = \underline{0.873}$$

PHASE 6: BEGIN WEIGHT: W_5
(DESCENT) END WEIGHT: W_6

FROM TABLE 2.1: $\frac{W_6}{W_5} = \underline{0.985}$

PHASE 7: BEGIN WEIGHT: W_6
(LANDING, END WEIGHT: W_7
TAXI,
SHUTDOWN) FROM TABLE 2.1;

$$\frac{W_7}{W_6} = \underline{0.995}$$

MISSION FUEL FRACTION, M_{ff}:

$$M_{ff} = \frac{W_1}{W_{TO}} \cdot \frac{W_2}{W_1} \cdot \frac{W_3}{W_2} \cdot \frac{W_4}{W_3} \cdot \frac{W_5}{W_4} \cdot \frac{W_6}{W_5} \cdot \frac{W_7}{W_6}$$

$$= (0.990)(0.995)(0.995)(0.970)(0.873)(0.985)(0.995)$$

$$M_{ff} = 0.813$$

STEP 3: (cont)

$$W_{F_{USED}} = (1 - M_{ff})\, W_{TO}$$

$$= (1 - 0.813)\, W_{TO}$$

$$W_{F_{USED}} = 0.187\, W_{TO}$$

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MISSION FUEL WEIGHT:

$$W_F = W_{F_{USED}} + W_{F_{RES}}$$

$$= W_{F_{USED}} + 0.25\, W_{F_{USED}}$$

$$= 1.25\, W_{F_{USED}}$$

$$W_F = 1.25\,(0.187\, W_{TO})$$

$$\underline{W_F = 0.234\, W_{TO}} \longleftarrow$$

STEP 4: CALCULATE A TENTATIVE VALUE FOR W_{OE} FROM:

$$W_{OE_{TENT}} = W_{TO_{GUESS}} - W_F - W_{PL}$$

$$W_{OE_{TENT}} = 96{,}000 \text{ lbs} - 0.234\,(96{,}000 \text{ lbs}) - 20{,}500 \text{ lbs}$$

$$\underline{W_{OE_{TENT}} = 53{,}036 \text{ lbs}}$$

STEP 5: CALCULATE A TENTATIVE VALUE FOR W_E AS:

$$W_{E_{TENT}} = W_{OE_{TENT}} - W_{tfo} - W_{CREW}$$

$$= 53{,}036 \text{ lbs} - (0.005\, W_{TO}) - 820 \text{ lbs}$$

$$= 52{,}216 - 0.005\,(96{,}000)$$

$$\underline{W_{E_{TENT}} = 51{,}736 \text{ lbs}}$$

STEP 6: FIND THE ALLOWABLE VALUE OF W_E FROM SECTION 2.5.

FROM FIG. 2.8, PG. 24, AT $W_{TO} = 96{,}000$ lbs,

$$\underline{W_{E_{All}} = 55{,}000 \text{ lbs.}}$$

NOT WITHIN THE ALLOWABLE TOLERANCE. MUST NOW ITERATE.

DUE TO THE LARGE ERROR BETWEEN THE PREDICTED EMPTY WEIGHT AND THE ALLOWABLE EMPTY WEIGHT, A NEW VALUE FOR W_E AND W_{TO} ARE CALCULATED USING THE EMPTY WEIGHT EQUATION FROM REFERENCE (1).

$$W_E = \frac{W_{COMP}}{W_{METAL}} \cdot \text{inv.} \log_{10} \left\{ (\log_{10} W_{TO} - A)/B \right\}$$

FROM TABLE 2.15 OF REFERENCE (1):

$A = 0.3774$
$B = 0.9647$

$$W_E = 0.90 \ \text{inv.} \log_{10} \left\{ \frac{(\log_{10} W_{TO} - 0.3774)}{0.9647} \right\}$$

W_{TO} (lbs)	W_E (ALLOWABLE) (lbs)
86,000	47,653
90,000	49,953
95,000	52,832
96,000	53,408
98,000	54,563
100,000	55,717
105,000	58,608

USING THE VALUE OF $W_{TO} = 105{,}000$ lbs IN THE FUEL FRACTION EQUATIONS CALCULATED PREVIOUSLY, YIELDS:

W_{TO}	105,000 (lbs)
$W_F = 0.234\, W_{TO}$	24,570 (lbs)
$W_{OE_{TENT}} = W_{TO} - W_F - 20{,}500$	59,930 (lbs)
$W_{E_{TENT}} = W_{OE_{TENT}} - 0.005\, W_{TO} - 820$	58,585 (lbs)

THE ERROR BETWEEN THE PREDICTED EMPTY WEIGHT AND THE ALLOWABLE EMPTY WEIGHT IS:

$$\%\ \text{ERROR} = \frac{58{,}608 - 58{,}585}{58{,}608} \times 100\% = 0.04\%$$

TABLE L.1 PRELIMINARY WEIGHTS AND WEIGHT SENSITIVITIES FOR THE 100 PASSENGER COMMUTER

WEIGHTS: Take-Off Weight, W_{TO} = 112,288 lbs

Operating Weight Empty, W_{OWE} = 67,422

Empty Weight, W_E = 66,041 lbs

Payload Weight, W_{PL} = 20,500 lbs

Mission Fuel Weight, W_F = 24,366 lbs

WEIGHT SENSITIVITIES:

Payload Weight: dW_{TO}/dW_{PL} = 5.9

Empty Weight: dW_{TO}/dW_E = 1.6

Specific Fuel Consumption: dW_{TO}/dC_p = 202,659 lb/lb/lb/hr

Propeller Efficiency: $dW_{TO}/d\eta_p$ = -95,369 lbs

Lift-to-Drag Ratio: $dW_{TO}/d(L/D)$ = -5,067 lbs

Range: dW /dR = 54.0 lb/nm

Wing Area: S = 1,604 ft

Wing Aspect Ratio: AR = 12

Take-Off Power: P_{TO} = 26,750 hp

Required Lift Coefficients:

Clean, $C_{L_{MAX}}$ = 1.32

Take-Off, $C_{L_{MAX_{TO}}}$ = 1.80

Landing, $C_{L_{MAX_L}}$ = 3.0

L.3 PRELIMINARY PERFORMANCE SIZING

This section presents the calculations used in the preliminary performance sizing. The methods used are those presented in Reference (1).

1. PRELIMINARY WEIGHT SIZING

THE PRELIMINARY WEIGHT SIZING OF THE NASA-100 WAS DONE WITH THE CADAE 3 PROGRAM ON THE HARRIS-300 COMPUTER.

ASSUMPTIONS: $W_{ARALL} / W_{ALUM.} = 0.95$

$W_{PL} = 20{,}500$ lbs

$W_{CREW} = 820$ lbs

USING THIS PROGRAM, THE CALCULATED WEIGHTS ARE:

$W_{TO} = 112{,}288$ lbs

$W_E = 66{,}321$ lbs

$W_F = 24{,}363$ lbs

2. TAKE-OFF WEIGHT SENSITIVITIES

THESE SENSITIVITIES WERE ALSO CALCULATED WITH THE CADAE 3 PROGRAM.

ASSUMPTIONS: $C_p = 0.40$ lb/lb/hr
$\eta_p = 0.85$
$L/D = 16.0$
$R = 1500$ nm

GROWTH FACTORS WERE CALCULATED AS:

DUE TO PAYLOAD WEIGHT: $\partial W_{TO} / \partial W_{PL} = 5.9$

DUE TO EMPTY WEIGHT: $\partial W_{TO} / \partial W_E = 1.6$

SENSITIVITIES WERE DERIVED AS:

$\partial W_{TO} / \partial C_p = 202{,}659$ lb/lb/lb/hr

$\partial W_{TO} / \partial \eta_p = -95{,}369$ lbs

$\partial W_{TO} / \partial (L/D) = -5{,}067$ lbs

$\partial W_{TO} / \partial R = 54.0$ lb/nm

3. PRELIMINARY PERFORMANCE SIZING

THE PRELIMINARY PERFORMANCE SIZING OF THE NASA-100 WAS DONE WITH THE CADAE 2 PROGRAM ON THE HARRIS-800 COMPUTER.

TAKE-OFF DISTANCE, LANDING DISTANCE, AND CRUISE SPEED PROVED TO BE THE CRITICAL PERFORMANCE CRITERIA. SEE FIG. 1.

ASSUME : $A = 12$

3.1 STALL SPEED REQUIREMENTS

AT $V_{S\,(FLAPS-UP)} = 120$ kts , $(W/S)_{TO} = 73.2$ psf

AT $V_{S\,(FLAPS-DOWN)} = 100$ kts , $(W/S)_{TO} = 88.1$ psf

3.2 DRAG POLARS

LOW SPEED, CLEAN: $C_D = 0.0629 + 0.0312 C_L^2$
TAKE-OFF, GEAR UP: $C_D = 0.0779 + 0.0332 C_L^2$
TAKE-OFF, GEAR DOWN: $C_D = 0.0979 + 0.0332 C_L^2$
LANDING, GEAR UP: $C_D = 0.0929 + 0.0332 C_L^2$
LANDING, GEAR DOWN: $C_D = 0.1129 + 0.0332 C_L^2$

3.3 TAKE-OFF DISTANCE

THE VALUES OF POWER LOADING CORRESPONDING TO VARIOUS WING LOADINGS AND MAXIMUM TAKE-OFF LIFT COEFFICIENTS IS FOUND IN TABLE 1.

3.4 LANDING DISTANCE

THE RELATION BETWEEN WING LOADING AND $C_{L_{MAX_L}}$ IS GIVEN AS:

$$(W/S)_{TO} = 23.40\, C_{L_{MAX_L}}$$

THE VALUES SATISFYING THIS RELATION ARE PRESENTED IN TABLE 2.

3.5 CLIMB REQUIREMENTS

ALL CLIMB REQUIREMENT VALUES OF POWER LOADING ARE PRESENTED IN TABLE 3.

TABLE 1 -- POWER LOADINGS TO MEET TAKE-OFF DISTANCE REQUIREMENTS

$(W/S)_{TO}$ psf	$C_{L_{MAX_{TO}}}$ = 1.00	1.50	2.00	2.50	3.00
20.0	13.5	20.2	26.9	33.7	40.4
40.0	6.7	10.1	13.5	16.8	20.2
60.0	4.5	6.7	9.0	11.2	13.5
80.0	3.4	5.0	6.7	8.4	10.1
100.0	2.7	4.0	5.4	6.7	8.1

TABLE 2 -- MAXIMUM TAKE-OFF WING LOADINGS TO MEET LANDING DISTANCE REQUIREMENTS

$C_{L_{MAX_L}}$	$(W/S)_{TO\,MAX}$ psf
1.00	23.40
1.50	35.11
2.00	46.81
2.50	58.51
3.00	70.21

TABLE 3 -- CLIMB REQUIREMENT POWER LOADINGS AT A=12.

$(W/S)_{TO}$ psf	(W/P) lbs/hp	(W/P) lbs/hp	(W/P) lbs/hp	(W/P) lbs/hp	(W/P) lbs/hp	(W/P) lbs/hp
20.0	43.07	41.73	39.00	40.01	28.59	27.58
40.0	30.45	29.51	27.58	28.29	20.22	19.50
60.0	24.86	24.09	22.52	23.10	16.51	15.92
80.0	21.53	20.86	19.50	20.00	14.29	13.79
100.0	19.26	18.66	17.44	17.89	12.79	12.33
	- FAR 25.111 - INTIAL CLIMB SEGMENT (OEI)	- FAR 25.121 - TRANSITION SEGMENT CLIMB (OEI)	FAR 25.121 - SECOND SEGMENT CLIMB (OEI)	FAR 25.121 EN-ROUTE CLIMB SEGMENT (OEI)	FAR 25.119 LANDING CLIMB SEGMENT (AEO)	FAR 25.121 BALKED LANDING SEGMENT (OEI)

3.6 MANEUVERING REQUIREMENTS

THE CALCULATED RELATION IS AS FOLLOWS:

$$(W/P) = (W/S)/0.05 + 4646.88\,(W/S)$$

THE VALUES SATISFYING THIS RELATION ARE GIVEN IN TABLE 4.

3.7 CRUISE SPEED REQUIREMENTS

THE CALCULATED RELATION IS:

$$(W/P) = (W/S)/8.15$$

THE VALUES SATISFYING THIS RELATION ARE GIVEN IN TABLE 4.

3.8 MATCHING SIZING REQUIREMENTS

POINT P_3, FIG. 1, WAS CHOSEN AS THE POINT OF MAXIMUM ALLOWABLE WING LOADING:

$$(W/S)_{TO_{MAX}} = 70 \text{ psf}$$

AT THIS WING LOADING, THE POWER LOADING IS:

$$(W/P)_{TO} = 4.2 \text{ lbs/hp}$$

THE POWER REQUIRED IS CALCULATED AS:

$$P_{REQD} = 26,736 \text{ hp}$$

THE WING AREA REQUIRED IS:

$$S = 1604 \text{ ft}^2$$

THE DIFFERENT REQUIREMENTS OF THE PERFORMANCE SIZING ARE PRESENTED GRAPHICALLY IN FIG. 1.

TABLE 4 -- POWER LOADINGS REQUIRED FOR MANEUVERING AND FOR CRUISE SPEED

$(W/S)_{TO\ MAX}$ psf	$(W/P)_{TO\ MAX}$ lbs/hp (STATIC)	$(W/P)_{TO\ MAX}$ lbs/hp (STATIC)
20.0	87.30	1.23
40.0	44.20	2.45
60.0	30.08	3.68
80.0	23.21	4.91
100.0	19.23	6.14
	MANEUVERING REQUIREMENT	CRUISE SPEED REQUIREMENT.

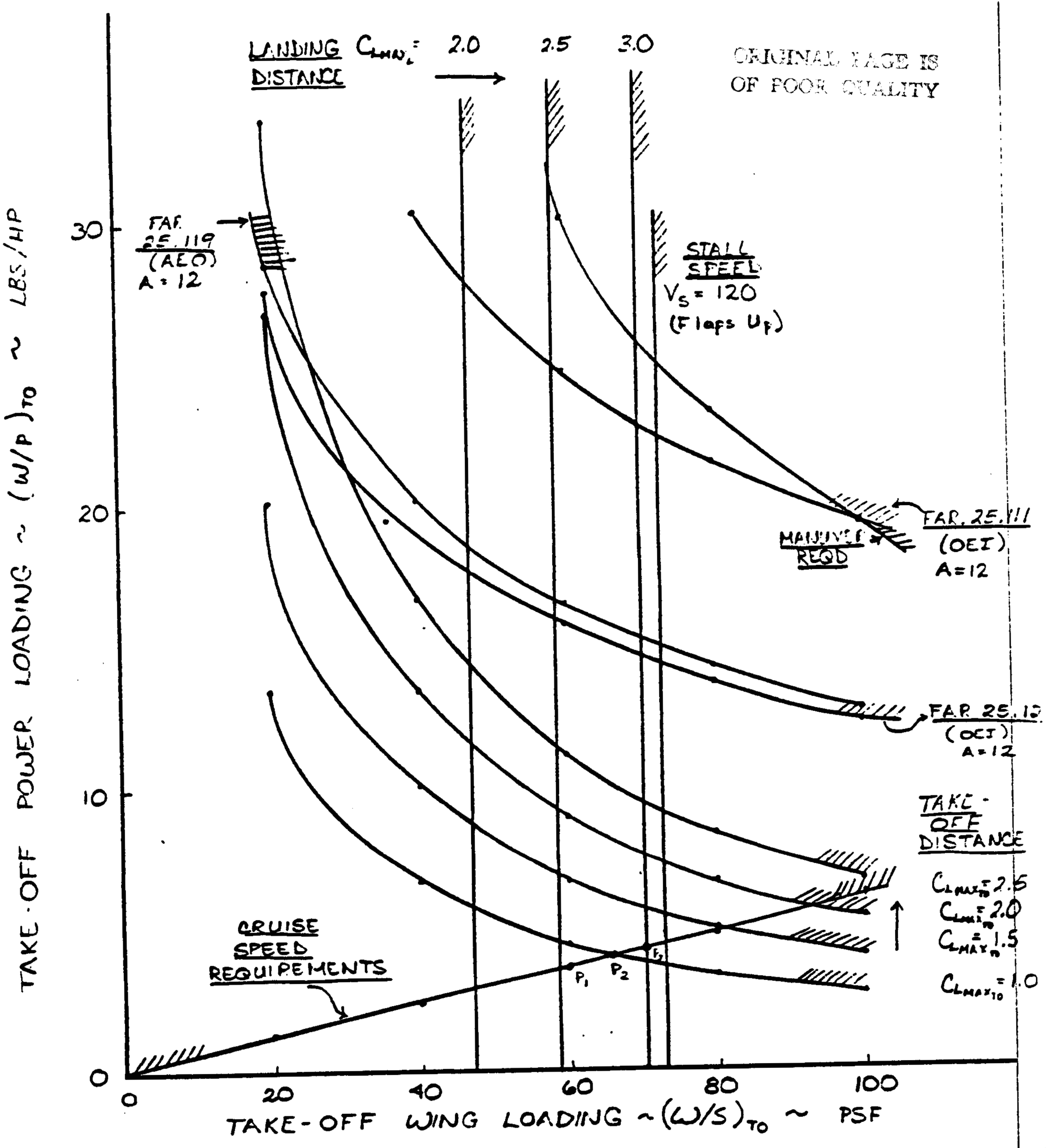

FIG. 1 - MATCHING RESULTS FOR 100 PASSENGER REGIONAL TURBO-PROP PERFORMANCE SIZING.

$\left(\frac{W}{P}\right)_{P_1} = 3.6$

$\left(\frac{W}{P}\right)_{P_2} : 4.0$

$\left(\frac{W}{P}\right)_{P_3} : 4.2$

L.4 FLAP SIZING

Using methods presented in Reference (2) the flap geometry required to provide the necesary incremental lift coefficients for take-off and landing were calculated. The results are presented in Table L.2.

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7.1:

$C_{L_{MAX}} = 1.5$

$C_{L_{MAX_{TO}}} = 1.8$

$C_{L_{MAX_L}} = 3.0$

7.2:

'LONG - COUPLED' ⇒ $l_h / \bar{c} > 5.0$

NEEDED: $C_{L_{MAX_W}} = 1.05 C_{L_{MAX}}$

$= 1.05(1.5)$

$C_{L_{MAX_W}} = 1.57$

$\cos \Lambda_{c/4} = .9744$

$C_{L_{MAX_W}}$ UNSWEPT $= C_{L_{MAX_W}}$ SWEPT $/ .9744$

$C_{L_{MAX_W}}$ UNSWEPT $= \underline{1.61}$

ALLOWED: $k_\lambda = 0.95$

$\lambda = 0.4$

$C_{L_{MAX_W}} = k_\lambda (C_{l_{max_r}} + C_{l_{max_t}})/2$

$C_{L_{MAX_W}} = \underline{1.42}$

WON'T WORK

NEEDED: $C_{L_{MAX_W}}$ UNSWEPT $= 1.42$

$C_{L_{MAX_W}}$ SWEPT $= 1.38$

$C_{L_{MAX}} = \underline{\underline{1.32}}$

NOW ASSUME $C_{l_{max}} = \underline{1.32}$

7.3: $\Delta C_{L_{MAX}}$ NEEDED:

TAKE-OFF:

$$\Delta C_{L_{MAX_{TO}}} = 1.05\,(1.8 - 1.32)$$
$$\Delta C_{L_{MAX_{TO}}} = \underline{0.504}$$

LANDING:

$$\Delta C_{L_{MAX_L}} = 1.05\,(3.0 - 1.32)$$
$$\Delta C_{L_{MAX_L}} = \underline{1.76}$$

7.4: $K_\Lambda = \underline{0.906} = (1 - 0.08\cos^2\Lambda_{c/4})/\cos^{3/4}\Lambda_{c/4}$

S_{WF}/S	S/S_{WF}	$\Delta C_{l_{max_{TO}}} = 0.457\,(S/S_{WF})$	$\Delta C_{l_{max_L}} = 1.59\,(S/S_{WF})$
0.2	5.00	2.28	7.95
0.3	3.33	1.52	5.29
0.4	2.50	1.14	3.97
0.5	2.00	0.914	3.18
0.6	1.67	0.763	2.65
0.7 ←	1.43	0.653 ←	2.27 ←
0.8	1.25	0.571	1.99
0.9	1.11	0.507	1.76

7.5: ASSUME: $C_{L_\alpha} = 2\pi$; FOWLER FLAPS;

$\delta_{F_{TO}} = 20$ deg $= 0.349$ rad; $\delta_{F_L} = 40$ deg $= 0.698$; $\Delta C_l = C_{l_{\alpha_f}}\,\alpha_{\delta_f}\,\delta_f$

OBTAINABLE VALUES:

c_f/c	K	$C_{l_{\alpha_f}}$	$\alpha_{\delta_{F_{TO}}}$	$\alpha_{\delta_{F_L}}$	$\Delta C_{l_{TO}}$	ΔC_{l_L}
0.25	0.97	7.85	0.49	0.40	1.34	2.19
0.30	0.94	8.17	0.54	0.42	1.54	2.40
0.35	0.83	8.48	0.56	0.46	1.66	2.72
	(Fig 7.4)	(Eqn. 7.17)	(Fig 7.3)	(Fig 7.3)	(Eqn. 7.9)	

7.5 (cont)

$\Delta C_l = (1/K)\, \Delta C_{l\,max}$

WHAT WE WANT (ΔC_l)

	$c_f/c = 0.25$; $K = 0.97$		$c_f/c = 0.3$; $K = 0.94$		$c_f/c = 0.35$; $K = 0.83$	
S/S_{WF}	$\Delta C_{l_{TO}}$	ΔC_{l_L}	$\Delta C_{l_{TO}}$	ΔC_{l_L}	$\Delta C_{l_{TO}}$	ΔC_{l_L}
5.00	2.35	8.20	2.42	8.46	2.75	9.58
3.33	1.57	5.45	1.62	5.63	1.83	6.37
2.50	1.17	4.09	1.21	4.22	1.37	4.78
2.00	0.942	3.28	0.972	3.38	1.10	3.83
1.67	0.787	2.73	0.811	2.82	0.919	3.19
1.43	0.673	2.34	0.695	2.41 ←	0.787	2.73 ←
1.25	0.589	2.05 ←	0.607	2.12	0.688	2.40
1.11	0.523	1.81	0.539	1.87	0.611	2.12

NO LEADING EDGE CALCULATIONS DONE.

RESULTS : CH. 7

$C_{L_{MAX}} = 1.32$

$C_{L_{MAX_{TO}}} = 1.80$

$C_{L_{MAX_L}} = 3.00$

$S_{WF}/S = 0.7$

$S/S_{WF} = 1.43$

$\Delta C_{l_{TO}} = 0.695$ ALLOWED: $\Delta C_{l_{TO}} = 1.54$

$\Delta C_{l_L} = 2.41$ ALLOWED: $\Delta C_{l_L} = 2.40$

$c_f/c = 0.30$

$\delta_{F_{TO}} = 20$ deg

$\delta_{F_L} = 40$ deg

FOWLER FLAPS

$C_{l_{\alpha_f}} = 2\pi$

$\Delta C_{l\,max_{TO}} = 0.653$ NEEDED: 0.504

$\Delta C_{l\,max_L} = 2.27$ NEEDED: 1.76

6.8: AILERON CHORD RATIO: 0.30
AILERON SPAN RATIO: 0.76 - 1.00 } SEE CH. 7 CALCULATIONS

(FUSELAGE: 0.0 to 0.76 span)

6.9: REAR SPAR LOCATION:

$$(1 - 0.30 - \underset{\text{(CLEARANCE)}}{0.005})c = 0.695c$$

FRONT SPAR: ASSUME: 0.20c

6.10: WING FUEL VOLUME:

ASSUME: NO FUEL BEYOND 0.85 span
ASSUME: DRY BAYS NEEDED.

$$\tau_w = (t/c)_t \,/\, (t/c)_r = 0.13/0.13 = 1.0$$

$$\lambda_w = 0.4 \quad \text{(GIVEN)}$$

$$(t/c)_r = 0.13$$

$$S = 1604 \text{ ft}^2$$

$$b = 138.7 \text{ ft}$$

$$0.85\, b = 118 \text{ ft}$$

$$V_{WF} = 0.54 \frac{(1604)^2}{138.7} (0.13) \cdot \left\{(1+0.4+0.16)/(1.+0.4)^2\right\}$$

$$\underline{V_{WF} = 1036 \text{ ft}^2}$$

NEEDED: $V_{WF} = W_F \,/\, 49.0 \text{ lbs/ft}^3$ (JP-4 L&R)

$= 24{,}363 \text{ lbs} \,/\, 49.0$

$$\underline{V_{WF} = 497 \text{ ft}^3}$$

TABLE L.2 100 PASSENGER FLAP GEOMETRY

Trailing Edge Fowler Flaps

$C_f/c = 0.30$

$S_{wf}/S = 0.70$

$b_f/b = 0.76$

$\delta_{f_{TO}} = 20$ deg

$\delta_{f_L} = 40$ deg

L.5 CLASS I EMPENNAGE SIZING

This section presents the sizing of a convintional T-tail empennage using the $\overline{V}$-bar method presented in Reference (2).

8.1 CONFIGURATION: T-TAIL

WITH ENGINE IN THE MIDDLE OF THE VERTICAL TAIL.

NEED 15° CLEARENCE FOR TAKE-OFF

3-ENGINE CONFIG: TURBO-PROP

8.2 DISPOSITION OF THE EMPENNAGE.

ASSUME: $X_V = 49.0$ Ft

$X_h = 58.0$ Ft

8.3 DETERMINATION OF THE EMPENNAGE SIZE.

GIVEN: $\bar{V}_H = 1.08$

$S_e/S_H = 0.36$

$\bar{V}_V = 0.083$

$S_r/S_V = 0.34$

$S_a/S = 0.06$

$$S_h = \bar{V}_h S \bar{c} / X_h \qquad (8.3)$$

$$S_V = \bar{V}_V S b / X_V \qquad (8.4)$$

FROM PRELIMINARY SIZING: (PAGE 3)

$b = 139$ Ft

$\bar{c} = 11.6$ Ft

$S = 1604$ Ft

THUS:

$$S_h = (1.08)(1604)(11.6)/(58)$$

$$\underline{\underline{S_h = 347 \text{ Ft}^2}}$$

$$S_v = (0.083)(1604)(139)/(49)$$

$$\underline{\underline{S_v = 378 \text{ Ft}^2}}$$

8.5 DETERMINATION OF RUDDER AREA:

$$(S_r/S_v)(S_v)$$

$$S_r = (0.34)(378) = \underline{129 \text{ Ft}^2}$$

ELEVATOR AREA:

$$S_e = (0.36)(347) = \underline{125 \text{ Ft}^2}$$

L.6 CLASS I LANDING GEAR DESIGN

From chapter 9 in Reference (2) it was decided to choose a 30 inch diameter tire 9 inches wide. This tire can carry a 20,000 pound load.

From weight and balance calculations longitudinal gear placement criterion were met. There is 15 degrees between the main gear ground contact point and the forward c.g.

Figure L.1 shows that the lateral tip-over criterion is met for a 228 inch wheel base.

CH 9:

9.1: CHOICE: RETRACTABLE LANDING GEAR

9.2: CHOICE: TRICYCLE CONFIGURATION

9.3: GO TO CHAP. 10.

CHAP 10: CLASS I WEIGHT AND BALANCE

10.1: CLASS I COMPONENT WEIGHT BREAKDOWN

			W (lbs)	
1.	FUSELAGE GROUP	=	14,260	
2.	WING GROUP	=	13,043	
3.	EMPENNAGE GROUP	=	3,256	
4.	ENGINE GROUP	=	14,260	
5.	LANDING GEAR GROUP	=	4,828	
6.	FIXED EQUIPMENT GROUP	=	16,394	
7.	TRAPPED FUEL & OIL	=	561	
8.	CREW	=	820	
9.	FUEL	=	24,366	
10.	PASSENGERS	=	17,500	> 20,500
11.	BAGGAGE	=	3,000	
12.	CARGO	=	0	
13.	MILITARY LOAD	=	0	
		W_{TO}	112,288	= 112,233 ✓checks

$$W_E = \sum_{i=1}^{6} W_i = \begin{array}{r} 14,260 \\ 13,043 \\ 3,256 \\ 14,260 \\ 4,828 \\ \underline{16,394} \\ 66,041 \end{array} \neq 66,321 \text{ lbs}$$

$$W_{OE} = \sum_{i=1}^{8} W_i = \begin{array}{r} 66,041 \\ 561 \\ \underline{820} \\ 67,422 \end{array}$$

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10.2: SEE DRAWING 1.

9.5: MAXIMUM STATIC LOAD PER STRUT:

(9.1) $P_n = (W_{TO} l_m) / (l_m + l_n)$ (NOSE-WHEEL STRUT)

$P_m = (W_{TO} l_n) / n_s (l_m + l_n)$ (MAIN-GEAR STRUT)

$cg_{W_{TO}} = 957$ F.S.

$l_m = (1005 - 957)$ in $= 48$ in

$l_n = (957 - 246)$ in $= 711$ in

SCALED FROM DRAWING

$l_m + l_n = 711 + 48 = 759$ in

NOSE-WHEEL:

$P_n = (112,233 \text{ lbs})(48 \text{ in}) / (759 \text{ in.})$

$P_n = 7101$ lbs

MAIN-GEAR:

$P_m n_s = (112,233 \text{ lbs})(711 \text{ in}) / (759 \text{ in})$

$P_m n_s = 105,187$ lbs

$P_m / 2\text{struts} = 105,187 / 2 = 52,594$ lbs / STRUT

9.6: WHEELS PER STRUT: 2 (NOSE), 4 (MAIN)

9.7: COMPUTE: $\dfrac{P_n}{W_{TO}} = \dfrac{7101 \text{ lbs}}{112,283 \text{ lbs}} = 0.063$

$\dfrac{n_s P_m}{W_{TO}} = \dfrac{105,187}{112,233} = 0.937$

CORRELATES TO VALUES IN TABLE 9.2.

TIRES: 30" × 9" (10 TIRES)

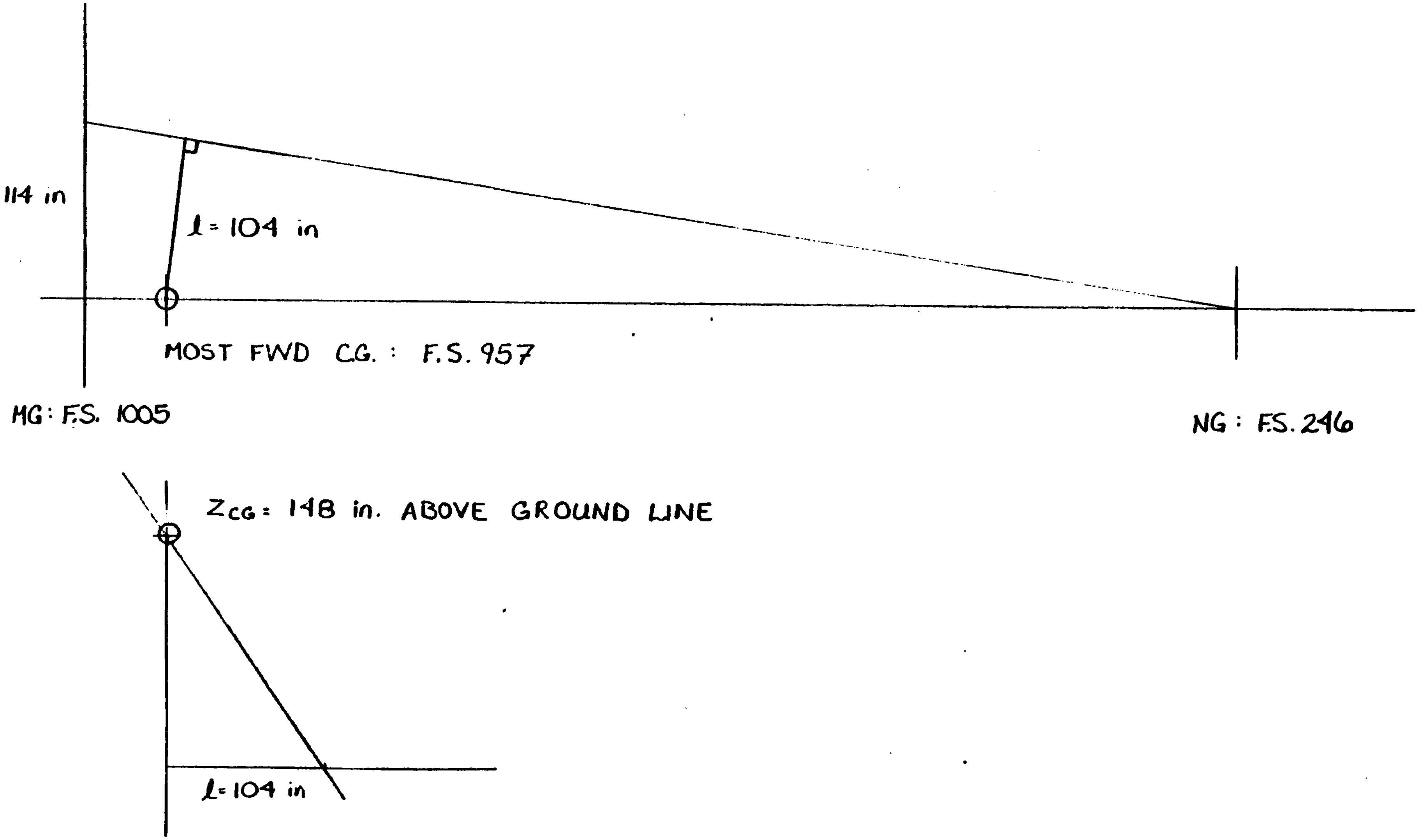

FIG L.1 -- TEST FOR LATERAL TIP-OVER CRITERIA

L.7 STABILITY AND CONTROL CALCULATIONS

Calculation of required stability derivatives are presented in this section.

100 PAX MULTHOPP INTEGRATION.

BODY SEGMENT	W_F	X_i	$W_F^2(X_i)$	ΔX_i
1	87	623	8,100	120
2	95	509	9,025	150
3	95	355	9,025	145
4	95	208	9,025	145
5	95	73.5	9,025	147
6	95	94.5	9,025	189
7	95	290	9,025	205
8	75	477	5,625	235
E(1)	37	225	1,369	150
E(2)	37	225	1,369	150

DETERMINATION OF $C_{L_{\alpha_W}}$ USING POLHAMUS:

$A = 12$; $\lambda = 0.4$; $\Lambda_{LE} = 15°$; $M = 0.70$

$$\beta = \sqrt{1-M^2} = 0.714$$

$$K = a_\infty / 2\pi / \beta$$

$$K = 6.01/2\pi/0.714 = 0.683$$

$$\mathcal{K} = 1 + \{(82 - 2.3\Lambda_{L.E}) - (0.22 - 0.153\Lambda_{L.E})A\}/100$$

$$\tan \Lambda_{LE} = \tan \Lambda_{c/2} + (2/A)(1-\lambda/1+\lambda)$$

$$\therefore \Lambda_{c/2} = 11° \quad ; \quad \mathcal{K} = 1.054$$

POLHAMUS GIVES:

$$\frac{K\,C_{L_{\alpha_w}}}{A} = \frac{2\pi}{2 + \sqrt{\frac{A^2\beta^2}{K^2}\left(1 + \frac{\tan^2\Lambda_{c/2}}{\beta^2}\right) + 4}}$$

$$\underline{\underline{C_{L_{\alpha_w}}}} = 4.72 \text{ RAD}^{-1} = 0.0824 \text{ deg}^{-1}$$

CORRECTION FACTOR FOR $d\bar{\varepsilon}/d\alpha$:

$$\left.\frac{d\bar{\varepsilon}}{d\alpha}\right|_{0.0737} = \left.\frac{d\bar{\varepsilon}}{d\alpha}\right|_{0.08} \times \frac{(0.0824)}{(0.08)}$$

$$\frac{d\bar{\varepsilon}}{d\alpha} = \left.\frac{d\bar{\varepsilon}}{d\alpha}\right|_{0.08} \times (1.03)$$

$d\bar{\varepsilon}/d\alpha$ FOR SEGMENTS 1-5:

SEGMENT	$d\bar{\varepsilon}/d\alpha\,\vert_{0.08}$	$d\bar{\varepsilon}/d\alpha\,\vert_{0.0824}$
1	1.00	1.03
2	1.00	1.03
3	1.10	1.13
4	1.18	1.22
5	2.00	2.06

$d\bar{\epsilon}/d\alpha$ FOR SEGMENTS 6 - E(2)

$l_H = 695$; $C_F = 200$

FROM FIG 3.25 IN THE 550 BOOK:

$d\epsilon/d\alpha = 0.167$.

FROM FIG 3.26 IN THE 550 BOOK, CORRECTION FACTOR = 0.970 ⇒ $(1 - d\epsilon/d\alpha) = 0.838$

SEGMENT	x_i/l_H	$d\bar{\epsilon}/d\alpha$
6	0.136	0.114
7	0.417	0.349
8	0.686	0.575
E(1)	0.324	0.272
E(2)	0.324	0.272

DETERMINATION OF $dM/d\alpha$:

$$\frac{dM}{d\alpha} = \frac{\bar{q}}{36.5} \sum_{i=1}^{2E} w_F^2(x_i) \left.\frac{d\bar{\epsilon}}{d\alpha}\right|_i \Delta x_i \quad (\deg^{-1})$$

$$\frac{dM}{d\alpha} = \frac{\bar{q}}{36.5}\Big[(579)+(807)+(956)+(924)+(1{,}582)+ +(113)+(374)+(440)+2(32)\Big]$$

$$\frac{dM}{d\alpha} = 157\,\bar{q}$$

$$\Delta \bar{X}_{ac_{BODY}} = \frac{-\frac{dM}{d\alpha}\ (\deg^{-1})}{\bar{q} S \bar{c}\, C_{L_{\alpha_W}}\ (\deg^{-1})}$$

$$\Delta \bar{X}_{ac_{BODY}} = \frac{-157\,\bar{q}}{\bar{q}(1604)(11.5)(0.0824)}$$

$$\underline{\underline{\Delta \bar{X}_{ac_{BODY}} = -0.103}}$$

ASSUME $\bar{X}_{ac_W} = 0.25$

$\therefore\ \bar{X}_{ac_{WB}} = 0.15$

CLASS I STABILITY AND CONTROL ANALYSIS.

STEP 11.1 LONGITUDINAL X-PLOT

$$\bar{X}_{ac_A} = \frac{\left[\bar{X}_{ac_{WB}} + \left\{C_{L_{\alpha_h}}(1 - d\varepsilon_h/d\alpha)(S_h/S)\,\bar{X}_{ac_h}\right\}/C_{L_{\alpha_{WB}}}\right]}{F}$$

WHERE $F = \left[1 + \left\{C_{L_{\alpha_h}}(1 - d\varepsilon_h/d\alpha)(S_h/S)\right\}/C_{L_{\alpha_{WB}}}\right]$

DETERMINATION OF $C_{L_{\alpha_h}}$:

$A = 5.3$; $\lambda = 0.35$; $\Lambda_{LE} = 29°$

$M = 0\ 70$, $\beta = 0.714$, $K = 0.683$

$K = 1 + \left\{(82 - 23\Lambda_{LE}) - (0.22 - 0.153\Lambda_{L.E.})A\right\}/100$

$\tan \Lambda_{L.E.} = \tan \Lambda_{c/2} + (2/A)(1 - \lambda/1 + \lambda)$

$\therefore \Lambda_{c/2} = 20.4°$; $K = 1.06$

$$\frac{K\,C_{L_{\alpha_h}}}{A} = \frac{2\pi}{2 + \sqrt{\frac{A^2\beta^2}{K^2}\left(1 + \frac{\tan^2 \Lambda_{c/2}}{\beta^2}\right) + 4}}$$

$\therefore \underline{\underline{C_{L_{\alpha_h}}}} = 3.67\ \mathrm{RAD}^{-1} = 0.0641\ \mathrm{deg}^{-1}$

$3.88\ \mathrm{RAD}^{-1}$ w/ $\lambda = 22°$

GIVEN $C_{L_{\alpha_h}} = 3.67$ RAD^{-1}, $C_{L_{\alpha_{WB}}} = 4.22$ RAD^{-1}

$\bar{X}_{ac_h} = 5.62$, $(1 - d\varepsilon/d\alpha) = 0.838$

$S_h = 347 \text{ Ft}^2$, $S = 1609 \text{ Ft}^2$

$\therefore F = 1 + 5.056 \times 10^{-4} S_h$

$$\therefore \bar{X}_{ac_A} = \frac{0.150 + 0.00255 S_h}{1 + 4.54 \times 10^{-4} S_h}$$

S_h	$\bar{X}_{ac_A}$	N_h (lbs)	$\bar{X}_{cg_{AFT}}$
100	0.387	617	0.435
120	0.432	740	0.442
130	0.455	802	0.449
150	0.500	925	0.457
170	0.542	1,049	0.464
190	0.584	1,172	0.471
210	0.626	1,296	0.486

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STATIC MARGIN:

$$dC_m / dC_L = \overline{X}_{cg} - \overline{X}_{ac} = -0.05 \qquad (11.4)$$

THE LONGITUDINAL X-PLOT GIVES:

$\overline{X}_{cg_{AFT}} = 0.456$, $\overline{X}_{ac} = 0.506$

$$\therefore \underline{\underline{dC_m / dC_L = -0.05 \quad \text{FOR} \quad S_h = 155 \text{ ft}^2}}$$

STATIC DIRECTIONAL STABILITY (DIRECTIONAL X-PLOT)

$$C_{n_\beta} = C_{n_{\beta_W}} + C_{n_{\beta_B}} + C_{L_{\alpha V}} (S_V/S)(X_V/b) \qquad (11.8)$$

ASSUME : $C_{n_{\beta_W}} = 0$

$$C_{n_{\beta_B}} = -57.3\, K_N K_{R_l} \frac{S_{B_S}}{S} \frac{l_B}{b} \quad (\text{RAD}^{-1}) \qquad (7.16)$$

$S_{B_S} \simeq 127{,}361 \text{ in}^2 = 884 \text{ ft}^2$

$l_B = 1520$ in , $h_1 = 97$ in

$X_m = 888$ in $\quad h_2 = 97$ in

$h = 97$ in ; $h = 97$ in

$\dfrac{X_m}{l_B} = 0.584$; $\dfrac{l_B^2}{S_{B_S}} = 18.1$

$\sqrt{\dfrac{h_1}{h_2}} = 1$; $\dfrac{h}{W} = 1$

FROM FIGURE 7.19:

$$K_N = 0.00085$$

$$R_{\ell_{FUSELAGE}} = \frac{V\ell}{\nu}$$

GIVEN: $V = 696\ 3$ ft/sec ; $\ell = 126.7$ ft

$\nu = 3.50 \times 10^{-4}$ ft²/sec

$$\therefore \quad R_{\ell_{FUSELAGE}} = 2.52 \times 10^{8}$$

FROM FIGURE (7.20)

$$K_{R_\ell} = 2.13$$

THUS: $C_{n_{\beta_B}} = (-57.3)(0\ 00085)(2.13)\left(\frac{884}{1604}\right)\left(\frac{126.7}{139}\right)$

$$\underline{\underline{C_{n_{\beta_B}} = -0.0521 \text{ RAD}^{-1}}} = 0.0091 \text{ DEG}^{-1}$$

USING POLHAMUS FOR $C_{L_{\alpha_V}}$:

$A = 1.4$, $\lambda = 0.6$, $\Lambda_{LE} = 45^{\circ}$; $\beta = 10$

$K = 0.956$

$K = 1 + \{(82 - 2.3\Lambda_{LE}) - (0\ 22 - 0.153\Lambda_{LE})A\}/100$

$\tan \Lambda_{L.E} = \tan \Lambda_{c/2} + (2/A)(1 - \lambda/1 + \lambda)$

$\therefore \quad \Lambda_{c/2} = 39.4^\circ \quad \& \quad K = 1.061$

$$\frac{K\, C_{L_{\alpha_V}}}{A} = \frac{2\pi}{2 + \sqrt{\frac{A^2\beta^2}{K^2}\left(1 + \frac{\tan^2 \Lambda_{c/2}}{\beta^2}\right) + 4}}$$

$\therefore \quad C_{L_{\alpha_V}} = 1.66 \text{ RAD}^{-1} = 0.0290 \text{ deg}^{-1}$

THUS: $C_{n_\beta} = (-0.0521 \text{ RAD}^{-1}) + (0.0290 \text{ deg}^{-1})(S_V/S)(X_V/b)$

WHERE: $X_V = 635$ in ; $b = 139$ ft $= 1{,}668$ in

$\therefore \quad C_{n_\beta} = (-9.093 \times 10^{-4} \text{ deg}^{-1}) + (0.0110 \text{ deg}^{-1})(S_V/S)$

USING $S = 1609 \text{ ft}^2$ YIELDS:

S_V (FT^2)	C_{n_β} (DEG^{-1})
150	0.000123
200	0.000467
250	0.000811
270	0.000949
280	0.001018
290	0.001087

FROM THE DIRECTIONAL X-PLOT THE REQUIRED VERTICAL TAIL AREA NEEDED FOR $C_{n_\beta} = 0.0010$ deg^{-1} IS FOUND TO BE

$$\underline{S_V : 277 \text{ ft}^2}$$

MINIMUM CONTROL SPEED WITH ONE ENGINE INOPERATIVE.

STEP 11.12 DETERMINATION OF THE CRITICAL ENGINE-OUT YAWING MOMENT:

$$N_{t_{crit}} = T_{TO_e} \, Y_t$$

$$Y_t = 60 \text{ in}$$

$$T_{TO_e} = \frac{550 \, \eta_p \, BHP}{V(1.1)}$$

WHERE: $V = 181$ ft/sec ; $\eta_p = 0.85$; $BHP = 13,350$

$$T_{TO_e} = 31,365 \text{ lbs}$$

THUS: $N_{t_{crit}} = (31,365)(60 \text{ in}/12) = 156,825$ ft·lb

$$N_D = 0.25 \, N_{t_{crit}} = 39,206 \text{ ft·lb}$$

$$V_{mc} = 1.2 \, V_s = 1.2(140 \text{ ft/sec}) = 168 \text{ ft/sec}$$

$$\delta_r = (N_D + N_{t_{crit}}) / \bar{q}_{mc} \, S b \, C_{n_{\delta_r}}$$

DETERMINATION OF $C_{n_{\delta_r}}$:

FROM: METHODS FOR ESTIMATING STABILITY AND CONTROL DERIVATIVES OF CONVENTIONAL SUBSONIC AIRPLANES.

$$C_{n_{\delta_r}} = -C_{y_{\delta_r}} \left(\frac{l_v \cos\alpha + z_v \sin\alpha}{b} \right)$$

WHERE: $\alpha = 0°$, $l_v = 585$ in , $b = 1{,}668$ in

$$C_{y_{\delta_r}} = C_{L_{\alpha_v}} \left[\frac{(\alpha_\delta)_{C_L}}{(\alpha_\delta)_{C_l}} \right] (\alpha_\delta)_{C_l} \, K' K_b \, \frac{S_v}{S}$$

WHERE: $C_{L_{\alpha_v}} = 0.0304 \text{ deg}^{-1}$; $(\alpha_\delta)_{C_l} = -0.73$

$\left[\frac{(\alpha_\delta)_{C_L}}{(\alpha_\delta)_{C_l}} \right] = 1.1$; $K' = 0.65$; $K_b = 1.0$

$S_v = 190 \text{ Ft}^2$; $S = 1604 \text{ Ft}^2$

$\therefore \quad C_{y_{\delta_r}} = -9.89 \times 10^{-6} \, S_v \text{ deg}^{-1}$

$\therefore \quad \underline{\underline{C_{n_{\delta_r}} = 3.470 \times 10^{-6} \, S_v \text{ deg}^{-1}}}$

$\bar{q}_{mc} = \frac{1}{2}(0.002377)(168 \text{ Ft/SEC})^2 = 33.5 \text{ lb/Ft}^2$

THUS: $25° = \dfrac{(156{,}825 + 39{,}206)}{(33.5)(1604)(139)(3.47 \times 10^{-6} \, S_v \text{ deg}^{-1})}$

$\therefore \quad \underline{\underline{S_v = 303 \text{ Ft}^2}}$

L.8 CALCULATION OF CLASS I DRAG POLARS

This section computes the airplane wetted area and estimates the skin friction drag. Class I drag polars are compared with the polars computed for the performance sizing. Table L.3 contains the drag polars and table L.4 contains a wetted area breakdown.

12.1. AIRPLANE COMPONENTS CONTRIBUTING TO WETTED AREA:

1. FUSELAGE
2. WING
3. EMPENNAGE
4. NACELLE

WETTED AREA FOR PLANFORM:

(12.1) $$S_{WET_{PLF}} = 2 S_{exp.\,plf.} \left\{ 1 + 0.25 (t/c)_r (1 + \tau\lambda) / (1 + \lambda) \right\}$$

$$\tau = 1.0$$
$$\lambda = 0.4$$
$$t/c = 0.13$$

$$S_{exp.\,plf} = 1604 \text{ sq ft} - \frac{17{,}700 \text{ sq in}}{144 \text{ sq in/sq ft}}$$
$$= 1604 - 123$$
$$S_{exp.\,plf.} = 1431 \text{ sq. ft.}$$

$$S_{WET_{PLF}} = 2(1481) \left\{ 1 + 0.25 (0.13) (1 \cancel{+ 0.4}) / (\cancel{1 + 0.4}) \right\}$$

$$\underline{S_{WET_{PLF}} = 3058 \text{ sq. ft}}$$

WETTED AREA FOR FUSELAGE:

(12.3) $$S_{WET_{FUS}} = \pi D_f l_f \left(1 - 2/\lambda_f\right)^{2/3} \left(1 + 1/\lambda_f^2\right)$$

$$D_f = 96.6 \text{ in} = 8.05 \text{ ft}$$

$$l_f = 1520 \text{ in} = 126.7 \text{ ft}$$

$$\lambda_f = l_f / D_f = 1520/96.6 = 15.74$$

$$S_{WET_{FUS}} = \pi (8.05)(126.7)\left(1 - 2/15.74\right)^{2/3} \left(1 + 1/15.74^2\right)$$
$$= \pi (8.05)(126.7)(0.913)(1.004)$$

$$\underline{S_{WET_{FUS}} = 2937 \text{ sq ft}}$$

WETTED AREA FOR NACELLES:

GIVEN:

$S_{WET_{NAC.}} = 124$ sq ft / Engine Installation

WETTED AREA FOR ENGINE PYLONS

$$S_{WET_{PYL.}} = (2)(2)\left[\frac{(67+87)}{2}(130)\right] = 40040 \text{ sq in}$$

$S_{WET_{PYL.}} = 278$ sq. ft

WETTED AREA FOR EMPENNAGE:

$$(12.1) \quad S_{WET_H} = 2 S_{exp\ plf}\left\{1 + 0.25(t/c)_r(1+\tau\lambda)/(1+\lambda)\right\}$$

$\tau = 1.0$

$t/c = 0.13$

$\lambda_H = 0.35$

$S_{exp\ plf} = 155$ sq. ft

$$S_{WET_H} = 2(155)\left\{1 + 0.25(0.13)\cancel{(1+0.35)}/\cancel{(1+0.35)}\right\}$$

$S_{WET_H} = 320$ sq. ft

$$12.1 \quad S_{WET_V} = 2 S_{exp\ plf}\left\{1 + 0.25(t/c)_r(1+\tau\lambda)/(1+\lambda)\right\}$$

$S_{exp\ plf} = \overset{303}{\cancel{190}}$ sq. ft

$(t/c)_r = 0.13$

$\tau = 1.0$

$\lambda = 0.6$

$$= 2(\overset{303}{\cancel{190}})\left\{1 + 0.25(0.13)\cancel{(1+0.6)}/\cancel{(1+0.6)}\right\}$$

$S_{WET_V} =$ ~~392~~ 626 sq. ft.

$S_{WET_{TOTAL}} = (3053 + 2937 + 248 + 278 + 320 + \overset{626}{\cancel{392}})$ sq. ft.

$S_{WET_{TOTAL}} = \overset{7467}{\cancel{7223}}$ sq. ft.

FROM FIG 3.22b) (PART I), RANGE IS $\{6000\ ft^2$ to $7300\ ft^2\}$.

THE VALUE FOUND FOR S_{WET} LIES INSIDE THIS RANGE.

12.2: EQUIVALENT PARASITE AREA:

USE FIG 3.21b(PART I), AT S_{WET} = ~~7233~~ 7467 ft^2,

f = ~~17.2~~ 18.5 ft^2

(ASSUMING: $C_f = 0.0025$)

12.3: FIND C_{D_0}.

C_{D_0} = f/S = (~~17.2~~ 18.5 ft^2)/(1604 ft^2)

C_{D_0} = ~~0.0107~~ 0.0115

e = 0.85 (ASSUMED)

$C_{D_{0_{M=.7}}}$ = 0.0107 + 0.0004 = ~~0.0111~~ 0.0119

12.4: COMPRESSIBILITY DRAG INCREMENT:

FROM FIG. 12.7, PART II :

$\Delta C_{D_{0_{COMP}}} = 0.0004$

12.5: FLAP DRAG INCREMENTS: (TAKE-OFF AND LAND)

ASSUMED: $\Delta C_{D_{0_L}} = 0.075$; $e = 0.80$

$\Delta C_{D_{0_{TO}}} = 0.015$; $e = 0.80$

12.6: LANDING GEAR DRAG INCREMENT:

ASSUMED $\Delta C_{D_{0_G}} = 0.020$

DRAG POLAR CALCULATIONS:

$$C_D = C_{D_0} + C_L^2/(\pi A e) \; ; \quad (L/D)_{max} = 0.5(\pi A e/C_{D_0})^{1/2}$$

$A = 12$

$C_{D_0\,(CLEAN)} = 0.0111 \quad ; \quad e = 0.85$

DRAG INCREMENTS:

$\Delta C_{D_{0_{TO}}} = 0.015 \quad ; \quad e = 0.80$

$\Delta C_{D_{0_L}} = 0.075 \quad ; \quad e = 0.80$

$\Delta C_{D_{0_G}} = 0.020$

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LOW SPEED, CLEAN:

$$C_D = 0.0119 + C_L^2/(\pi 12 \cdot .85) = 0.0119 + 0.0312\, C_L^2$$

$$(L/D)_{max} = 0.5(\pi 12 \cdot .85/0.0119)^{1/2} = \cancel{26.8} \;\; 25.9$$

$\underline{(L/D)_{cr=25} \;\; 20.4}$

TAKE-OFF FLAPS, GEAR UP:

$$C_D = (0.0119 + 0.015) + C_L^2/\pi 12(.80)$$
$$C_D = 0.0269 + 0.0332\, C_L^2$$

$$(L/D)_{MAX} = 0.5\,(\pi 12 \cdot .80/.0269)^{1/2} = \cancel{17.0} \;\; 16.7$$

TAKE-OFF FLAPS, GEAR DOWN:

$$C_D = 0.0269 + 0.020 + 0.0332 C_L^2 = 0.0469 + 0.0332 C_L^2$$

$$(L/D)_{MAX} = 0.5\,(30.16/0.0469)^{1/2} = \cancel{12.8} \;\; 12.7$$

LANDING FLAPS, GEAR UP:

$$C_D = 0.0119 + 0.075 + 0.0332 C_L^2 = 0.0869 + 0.0332 C_L^2$$

$$(L/D)_{MAX} = 0.5\,(30.16/.0369)^{1/2} = 9.3$$

LANDING FLAPS, GEAR DOWN:

$$C_D = 0.0869 + 0.020 + 0.0332 C_L^2 = 0.1069 + 0.0332 C_L^2$$

$$(L/D)_{MAX} = 0.5\,(30.16/.1069)^{1/2} = 8.4$$

TABLE L.3 -- Initial Drag Polars for NASA-100

PRELIMINARY SIZING RESULTS (BASED ON S = 500 ft^2)	DRAG POLAR	$(L/D)_{MAX}$
1. LOW SPEED, CLEAN:	$0.0629 + 0.0312 C_L^2$	11.28
2. TAKE-OFF, GEAR UP:	$0.0779 + 0.0332 C_L^2$	9.83
3. TAKE-OFF, GEAR DOWN:	$0.0979 + 0.0332 C_L^2$	8.77
4. LANDING, GEAR UP:	$0.0929 + 0.0332 C_L^2$	9.01
5. LANDING, GEAR DOWN:	$0.1129 + 0.0332 C_L^2$	8.17

CLASS I RESULTS ($S = 1604$ ft^2)	C_D: DRAG POLAR	$(L/D)_{MAX}$	$(L/D)_{cr}$
1. LOW SPEED, CLEAN:	$0.0111 + 0.0312 C_L^2$	26.8	21.6
2. TAKE-OFF, GEAR UP:	$0.0261 + 0.0332 C_L^2$	17.0	
3. TAKE-OFF, GEAR DOWN:	$0.0461 + 0.0332 C_L^2$	12.8	
4. LANDING, GEAR UP:	$0.0861 + 0.0332 C_L^2$	9.3	
5. LANDING, GEAR DOWN:	$0.1061 + 0.0332 C_L^2$	8.4	

PRELIMINARY (BASED ON 1604 ft^2)	C_D: DRAG POLAR	$(L/D)_{MAX}$	$(L/D)_{cr}$
1.	$0.0196 + 0.0312 C_L^2$	20.2	13.4
2.	$0.0346 + 0.0332 C_L^2$	14.7	
3.	$0.0546 + 0.0332 C_L^2$	11.7	
4.	$0.0946 + 0.0332 C_L^2$	8.9	
5.	$0.1146 + 0.0332 C_L^2$	8.1	

TABLE L.4 WETTED AREAS OF 100 PASSENGER AIRPLANE COMPONENTS

COMPONENT	WETTED AREA (ft^2)
Wing	3058
Horizontal Tail	320
Vertical Tail	626
Fuselage	2937
Engine Nacelles	124 x 2
Engine Pylons	278 x 2
Total	7467

APPENDIX M
ENGINEERING CALCULATIONS FOR THE 75 PASSENGER TWIN-BODY CONFIGURATION

TABLE OF CONTENTS

M.1 INTRODUCTION

The purpose of this appendix is to present the engineering calculations for the 75 passenger twin-body configuration. These calculations were used for the class I sizing of the airplane.

The stability and control calculations are contained in section M.2. The longitudinal and lateral-directional control surfaces are sized in this section. Section M.3 documents the class I drag polar calculations.

M.2 STABILITY AND CONTROL CALCULATIONS –

CENTER OF GRAVITY LOCATION:

$S_H : 2(100) = 200 \text{ ft}^2$

$W_H : 795.6 \text{ lbs}$

$W_H/S_H : 795.6/200 = 3.978 \text{ lb/ft}^2$

AFT C.G. LOCATION:

$W_i = 45308 \text{ lb}$

$W_i x_i = 27.557 \times 10^6 \text{ in-lb}$

$X_{cg} = 608.2 \text{ in}$

HORIZONTAL TAIL MOMENT ARM: 1045 in

S_H ft²	$S_{H_{TOTAL}}$ ft²	$X_{cg_{aft}}$	$\bar{X}_{cg_{aft}}$
60	120	605	.546
70	140	606	.554
80	160	607	.563
90	180	607	.572
100	200	608	.580
110	220	609	.589
120	240	610	.597
130	260	611	.606

AERODYNAMIC CENTER LOCATION:

FROM PREVIOUS STABILITY CALCULATIONS:

$\Delta \bar{x}_{ac_B} = -.39$

$\bar{x}_{ac_{WB}} = -.14$

$\bar{x}_{ac_{h_1}} = 5.77$

$C_{L_{\alpha_{h_1}}} = 3.45 \text{ rad}^{-1}$

$(1 - \frac{d\epsilon}{d\alpha})_1 = .764$

$\bar{X}_{ac_{h_2}} = 1.17$

$C_{L_{\alpha_{h_2}}} = 3.31 \text{ rad}^{-1}$

$\left(1 - \frac{d\epsilon}{d\alpha}\right)_2 = 1.00$

① ⇒ HORIZONTAL TAILS

② ⇒ ENGINE MOUNTING BAR

$$\bar{X}_{ac_A} = \frac{\left[\bar{X}_{ac_{WB}} + \left\{C_{L_{\alpha_{h_1}}}\left(1 - \frac{d\epsilon}{d\alpha}\right)\left(S_{h_1}/S\right)\bar{X}_{ac_{h_1}} + C_{L_{\alpha_{h_2}}}\left(1 - \frac{d\epsilon}{d\alpha}\right)_2\left(S_{h_2}/S\right)\bar{X}_{ac_{h_2}}\right\} / C_{L_{\alpha_{WB}}}\right]}{\left[1 + \left\{C_{L_{\alpha_{h_1}}}\left(1 - \frac{d\epsilon}{d\alpha}\right)_1\left(S_{h_1}/S\right) + C_{L_{\alpha_{h_2}}}\left(1 - \frac{d\epsilon}{d\alpha}\right)_2\left(S_{h_2}/S\right)\right\} / C_{L_{\alpha_{WB}}}\right]}$$

$$\bar{X}_{ac_A} = \frac{\left[-.14 + \left\{(3.45)(.764)\left(S_{h_1}/S\right)(5.77) + (3.31)(1.0)(.226)(1.17)\right\} / 4.985\right]}{\left[1 + \left\{(3.45)(.764)\left(S_{h_1}/S\right) + (3.31)(1.0)(.226)\right\} / 4.985\right]}$$

$$\bar{X}_{ac_A} = \frac{\left[-.14 + \left\{15.209\left(S_{h_1}/S\right) + .8752\right\} / 4.985\right]}{\left[1 + \left\{2.6358\left(S_{h_1}/S\right) + .7841\right\} / 4.985\right]}$$

S_H ft²	$S_{H\,TOTAL}$ ft²	$\bar{X}_{ac_A}$
60	120	.436
70	140	.498
80	160	.558
90	180	.618
100	200	.676
110	220	.732
120	240	.787
130	260	.842

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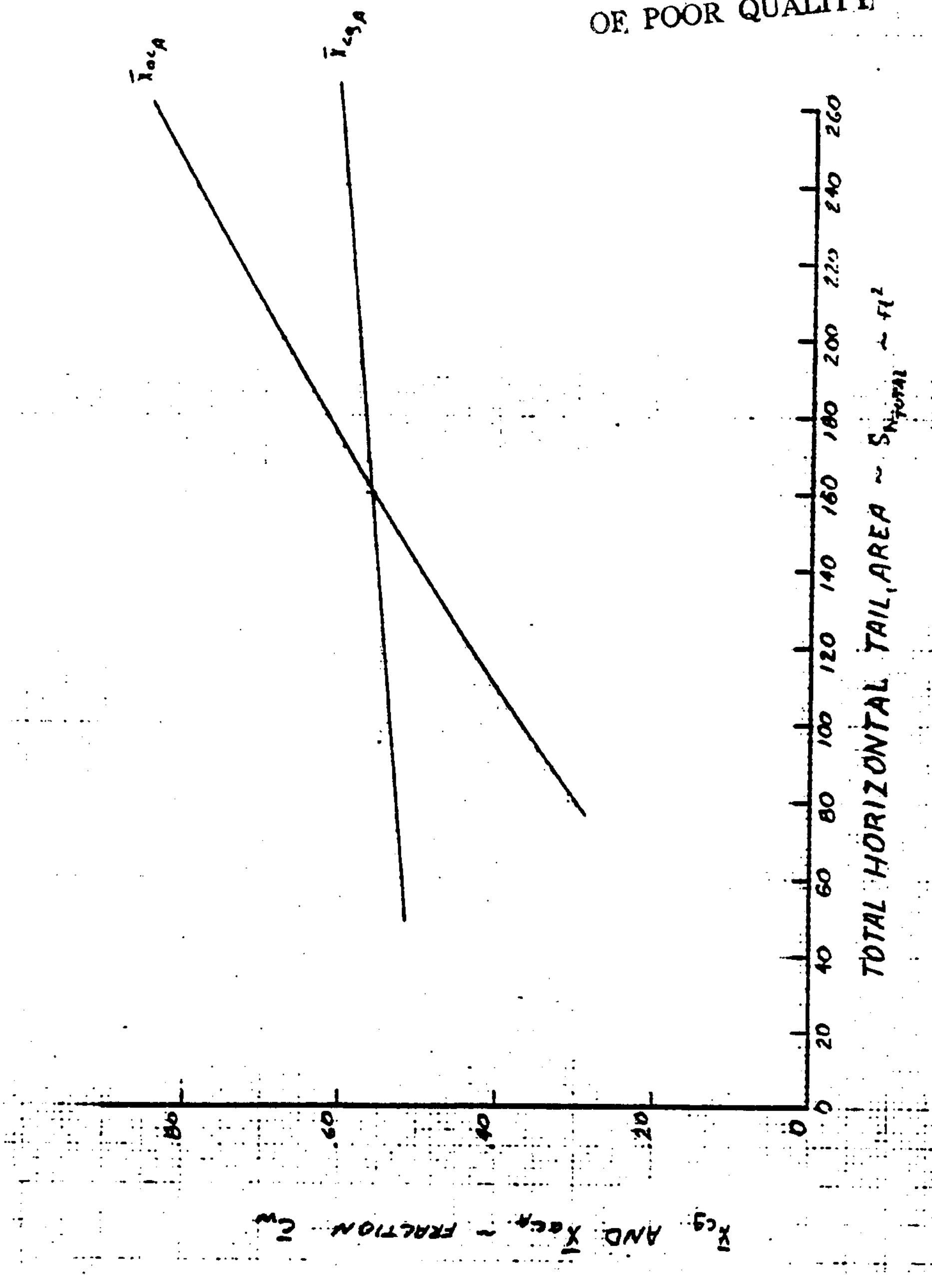

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STEP 2 - DIRECTIONAL STABILITY

CALCULATE LIFT CURVE SLOPE FOR THE VERTICAL TAIL - $C_{L_{\alpha_V}}$

GOEMETRIC PARAMETERS -

$S = 2 \times 130 = 260 \text{ FT}^2$ $\qquad \lambda = .3$

$\Lambda_{LE} = 58^\circ$ $\qquad A = 1.11$

$X_V = 34.67 \text{ FT}$

USING THE POLHAMUS EQUATION -

$\mathcal{K} = 1.0206$ $\qquad k = .682$

$\beta^2 = .51$ $\qquad \Lambda_{c/2} = 47^\circ$

$$C_{L_{\alpha_V}} = \frac{2\pi(1.11)}{\left(2 + \sqrt{\frac{(1.11)^2(.51)}{(.682)^2}\left(1 + \frac{(\tan 47)^2}{(.51)}\right) + 4}\right)1.0206}$$

$C_{L_{\alpha_V}} = 1.40 \text{ /RAD} = .0244 \text{ /DEG} \longleftarrow$ M = .70

CALCULATE $C_{n_{\beta_B}}$ -

USING THE METHOD OF REFERENCE 6, $C_{n_{\beta_B}}$ FOR THE SINGLE FUSELAGE WAS CALCULATED TO BE:

SINGLE BODY : $C_{n_{\beta_B}} = -.138$

FOR THE TWIN BODY, IT WILL BE ASSUMED THAT $C_{n_{\beta_B}}$ IS TWICE THAT OF THE SINGLE BODY -

TWIN BODY : $C_{n_{\beta_B}} = 2 \times (-.138) = -.276 \longleftarrow$

CALCULATE $C_{n_{\beta_V}}$ -

THE METHOD USED FOR THIS CALCULATION FOLLOWS THE PROCEDURE IN CHAPTER 7 OF REFERENCE 6 -

$$C_{Y_{\beta_V}} = -2 \left\{ \frac{C_{Y_{\beta_{V(WBH)}}}}{C_{Y_{\beta_{V_{eff}}}}} \right\} C_{Y_{\beta_{V_{eff}}}} \frac{S_V}{S}$$

USING FIGURE 7.9 OF REF. 6 -

$$b'_V / b_V = 1.0$$

$$A_{eff} / A = 1.5$$

$$A_{eff} = (1.5)(1.11) = 1.67$$

FROM FIGURE 7.8 OF REF. 6 -

$$C_{Y_{\beta_{eff}}} = 2.5$$

FROM FIGURE 7.10 OF REF. 6 -

$$b_H / l_B = .2$$

$$2r_1 / b_V = .28$$

$$\frac{C_{Y_{\beta_{V(WBH)}}}}{C_{Y_{\beta_{V_{eff}}}}} = .75$$

$$C_{Y_{\beta_V}} = -2(.75)(2.5)(S_V/722)$$

$$C_{Y_{\beta_V}} = -.0052\ S_V \ \text{rad}^{-1}$$

$$C_{n_{\beta_V}} = -C_{y_{\beta_V}} \left(l_v / b \right)$$

$$C_{n_{\beta_V}} = .0052\ S_V \left(29.2 / 104.5 \right)$$

$$C_{n_{\beta_V}} = .00145\ S_V$$

CALCULATE C_{n_β} -

$$C_{n_\beta} = C_{n_{\beta_V}} + C_{n_{\beta_B}}$$

$$C_{n_\beta} = .00145\ S_V - .276$$

$S_{V_{TOTAL}}$ (FT^2)	C_{n_β} (rad^{-1})	C_{n_β} $(deg)^{-1}$
180	-.014	-.000244
200	.015	.000262
220	.044	.000768
240	.073	.00127
260	.102	.00178
280	.131	.00229
300	.160	.00279

ENGINE OUT REQUIREMENTS -

$y_T = 3.9$ FT

$P_{TO_e} = 9000$ shp

$$T_{TO_e} = \frac{550\ bhp\ \eta_p}{V}$$

$$T_{TO} = \frac{(550)(9000)(.85)}{184}$$

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$$T_{TO_e} = 22867 \text{ lbs}$$

$$N_{t_{CRIT}} = (22867)(3.9) = \underline{89,181 \text{ ft-lbs}}$$

$$N_D = .25\, N_{t_{CRIT}}$$

$$N_D = (.25)(89,181) = \underline{22,295 \text{ ft-lbs}}$$

<u>CALCULATE</u> $\underline{C_{n_{\delta_R}} -}$

$$C_{y_{\delta_R}} = C_{L_{\alpha_V}} \frac{(\alpha_\delta)_{C_L}}{(\alpha_\delta)_{C_\ell}} (\alpha_\delta)_{C_\ell} \quad K' K_b \frac{S_V}{S}$$

$C_{L_{\alpha_V}} = 1.475$

$(\alpha_\delta)_{C_\ell} = -.7$

$\frac{(\alpha_\delta)_{C_L}}{(\alpha_\delta)_{C_\ell}} = 1.14$

$K_b = 1$

$K' = .65$

$$C_{y_{\delta_R}} = (1.475)(1.14)(-.7)(.65)(1.0)(S_V/722)$$

$$\underline{C_{y_{\delta_R}} = -.00106}$$

$$C_{n_{\delta_R}} = -C_{y_{\delta_R}} \frac{l_V}{b}$$

$$C_{n_{\delta_R}} = (.00106\, S_V)(29.2/109.5)$$

$$\underline{C_{n_{\delta_R}} = .000296\, S_V}$$

CALCULATE δ_R REQUIRED -

$$\delta_R = (N_D + N_{t_{CRIT}}) / \bar{q}_{mc} S b C_{n_{\delta_R}}$$

$$\bar{q} = (1/2)(.0023769)(184)^2 = 40.2 \text{ psf}$$

$$\delta_R = (22{,}295 + 89{,}181) / (40.2)(722)(109.5)(.00030\, S_v)$$

$$\delta_R = 124 / S_v \quad \text{rad}$$

$$\delta_R = 7112 / S_v \quad \text{deg}$$

$S_{v_{TOTAL}}$	$\delta_{R_{req}}$
160	44
180	40
200	36
220	32
240	30
260	27
280	25
300	24
320	22
340	21

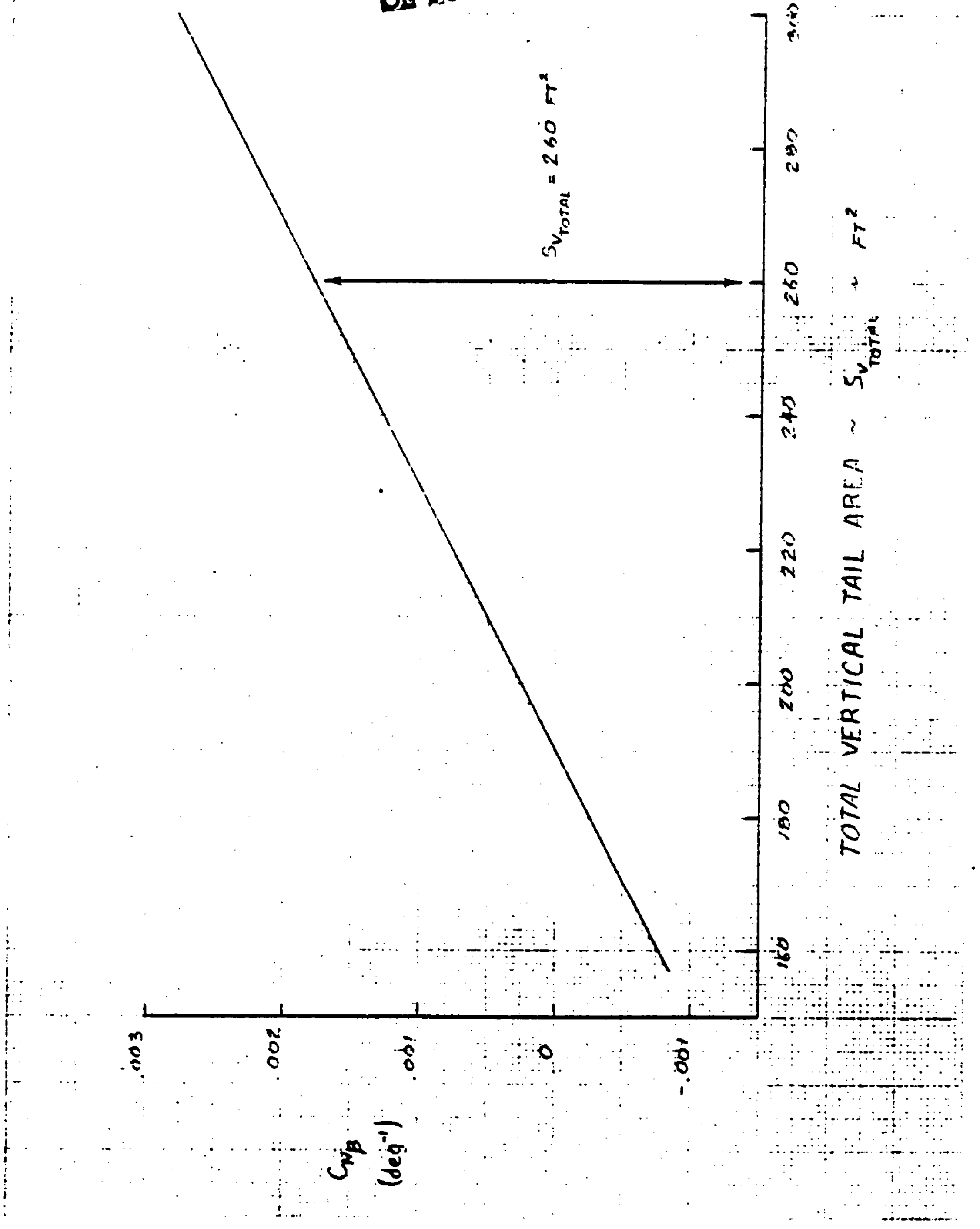

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FIGURE M.2 LATERAL-DIRECTIONAL X-PLOT FOR THE 75 PASSENGER TWIN-BODY

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M.3 CLASS I DRAG POLARS-

CALCULATE WETTED AREA-

WING-

$$S_{WET} = 2\, S_{exp} \{ 1 + 0.25 (t/c)_r\, (1 + \tau \lambda) / (1 + \lambda) \}$$

$$\tau = (t/c)_r / (t/c)_t$$

$$\lambda = c_t / c_r$$

FOR THE OUTBOARD WING SECTIONS-

$$S_{exp} = (32.7)(8.33)(1.4)$$

$$S_{exp} = 381.5 \text{ ft}^2$$

$$\tau = .13 / .10 = 1.3$$

$$\lambda = .4$$

$$S_{WET} = (2)(381.5) \{ 1 + .25(.13)(1 + (1.3)(.4)) / (1 + .4) \}$$

$$S_{WET} = 788 \text{ FT}^2$$

FOR THE INBOARD WING SECTION-

$$S_{exp} = (23.5)(8.75) = 206 \text{ FT}^2$$

$$\tau = 1.0$$

$$\lambda = 1$$

$$S_{WET} = (2)(206) \{ 1 + .25(.12)(1 + 1) / (1 + 1) \}$$

$$S_{WET} = 218 \text{ FT}^2$$

FOR THE TOTAL WING-

$$S_{WET} = 788 + 218 = 1006 \text{ FT}^2 \longleftarrow$$

HORIZONTAL TAIL -

$$S_{WET} = 2(102)\{1 + .25(.11)(1 + [1.1][.5])/(1+.5)\}$$

$$S_{WET} = 209.8 \ FT^2 \quad (\text{per H.T.})$$

$$S_{WET} = 2(209.8) = 419.6 \ FT^2 \longleftarrow$$

VERTICAL TAIL -

TWICE THE SINGLE BODY (36 PAX) VALUE:

$$S_{WET} = 2(267) = 534 \ FT^2 \longleftarrow$$

FUSELAGE -

TWICE THE SINGLE BODY (36 PAX) VALUE:

$$S_{WET} = 2(1702) = 3404 \ FT^2 \longleftarrow$$

NACELLE -

WETTED AREA WAS CALCULATED TO BE:

$$S_{WET} = 2(124) = 248 \ FT^2 \longleftarrow$$

PYLONS -

ENGINE - FUSELAGE SECTION -

$$\lambda = .83 \qquad \tau = 1$$

$$S_{EXP} = (6.25)(8.33)(1.83) = 95.3$$

$$S_{WET} = 2(95.3)\{1 + .25(.12)(1+.83)/(1+.83)\}$$

$$S_{WET} = 196 \ FT^2$$

CENTER SECTION -

$S_{exp} = (6.83)(6.25) = 42.7$

$S_{WET} = 2(42.7)\{1 + .25(.12)(2)/(2)\}$

$S_{WET} = 88.0 \ FT^2$

TOTAL PYLON AREA -

$S_{WET} = 2(196) + 88$

$S_{WET} = 480 \ FT^2$

COMPONENT	WETTED AREA
WING	1006
HORIZONTAL TAIL	420
VERTICAL TAIL	534
FUSELAGE	3404
NACELLES	248
PYLONS	480
TOTAL	6092

ASSUME: $C_f = .0025$

FROM FIG. 3.21 b) -

$f = 14.5$

$C_{D_0} = {}^{14.5}/_{722} = .0201$

C_{D_0} INCREMENTS -

COMPRESSIBILITY	.0002
LANDING GEAR	.0150
LANDING FLAPS	.0750

TAKE-OFF-

$C_{D_0} = .0201 + .015 = .0351$

$e = .80$
$A = 15.1$

$C_D = .0351 + .0264\ C_L^2$

CRUISE -

$C_{D_0} = .0201 + .0002 = .0203$

$e = .85$
$A = 15.1$

$C_D = .0203 + .0248\ C_L^2$

LANDING -

$C_{D_0} = .0201 + .0150 + .0750 = .1101$

$e = .80$
$A = 15.1$

$C_D = .1101 + .0264\ C_L^2$

CRUISE WEIGHT -

$W_{CR} = W_{TO} - .4\, W_F$

$W_{CR} : 60683 - .4(11240) = 56187$ lbs

LIFT COEFFICIENT -

$C_{L_{CR}} = (2)(56187) / (.0008897)(696.3)^2(722)$

$C_{L_{CR}} = .36$

$(L/D)_{CR} = .36/.0235$

$\underline{(L/D)_{CR} = 15.3} \longleftarrow$

IF A 10% REDUCTION IN C_{D_0} IS ASSUMED -

$\Delta C_{D_0} = (.0203)(.10) = .0020$

$C_D = .0183 + .0248\, C_L^2$

$(L/D)_{CR} = .36/.0215$

$\underline{(L/D)_{CR} = 16.7} \longleftarrow$

Appendix N:

100 Passenger Twin Body Design Calculations - Class 1 Summary

Table of Contents

The 50 passenger design is the basis for the 100 passenger twin body design. The preliminary weight and performance sizing was used for the twin body design but multiplied by a factor of 2. Also, the Class I sizing of the following components was used for the twin body design:

* cockpit and fuselage layouts
* wing planform design
* sizing and location of lateral control surfaces
* sizing high lift devices
* Empennage Sizing by V-bar method
* Landing gear sizing and disposition

For detailed calculations of the above, the 50 passenger design will have to be consulted.

N.2 Landing Gear Criterion

The following pages provides the research on the applicability of the 100 passenger twin body wide wheelbase arrangement.

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For the 100 passenger twin fuselage commuter transport an estimated wheelbase of 50 ft is required.

From Airport Engineering by Ashford and Wright, the following standards are given on runway and taxiway dimensions. See attached charts. (Pgs N -)

From the data compiled, the following conclusions can be made:

1. The design can operate out of any airline airport.

2. This design will not be able to operate [illegible] general aviation airports. General and basic transport general aviation airports have taxiway widths between 40 - 60 ft.

N.3 Class I Weight and Balance

The following pages give the Class I

- general arrangement drawing
- weight and balance calculations
- weight -c.g. excursion diagram

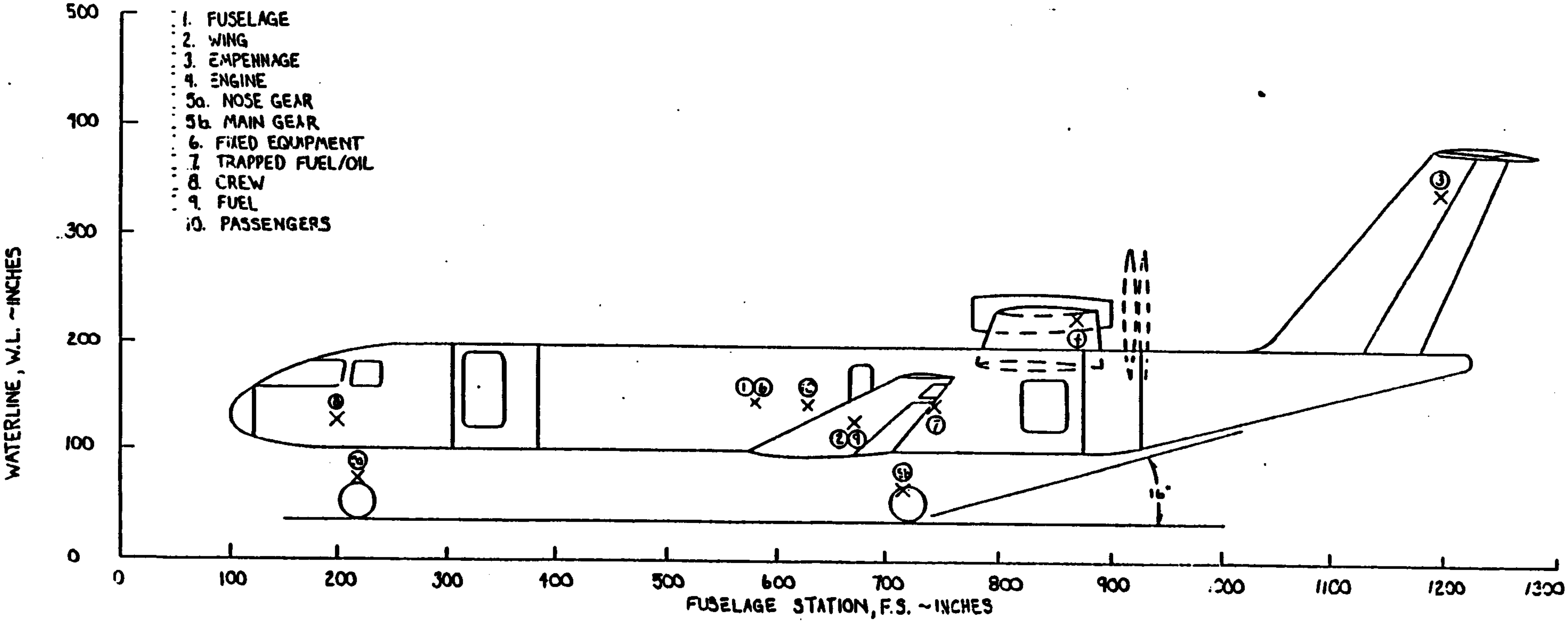

FIGURE 3.5.2. 100 PASSENGER TWIN BODY GENERAL ARRANGEMENT DRAWING.

Component	Weight lbs.	x in	z in		
uselage	10,704.00	578.00	148.00		
Wing	7,597.00	672.00	127.00		
mpennage	2,238.00	1,204.00	340.00		
ngine	8,470.00	870.00	222.00		
Nose Gear	746.00	220.00	74.00		
Main Gear	2,994.00	720.00	64.00		
ixed Equipment	12,354.00	578.00	148.00		
Empty Weight	45,103.00			xcg	683.24
				zcg	161.09
Trapped Fuel/Oil	420.00	745.00	178.00		
rew	615.00	200.00	120.00		
Operating Empty Weight	46,138.00			xcg	677.36
				zcg	160.69
uel	13,878.00	672.00	127.00		
Passengers	20,500.00	630.00	148.00		
ake-off Weight	80,516.00			xcg	664.38
				zcg	151.65

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Table 2.5.4 Twin Body 100 Passenger Commuter Class I Weight and Balance Calculation

No.	Component	Weight lbs	X_i in	Z_i in
1.	Fuselage	10704	578	148
2.	Wing	7597	672	127
3.	Empennage	2438	1204	340
4.	Engine	8470	870	222
5a.	Nose Gear	746	220	74
5b.	Main Gear	2994	720	64
6.	Fixed Eqpt.	12354	578	148
Empty Weight		$W_E = 45303$		$X_{cg_{We}} = 686$
				$Z_{cg_{We}} = 161$
7.	Trapped Fuel and Oil	420	745	178
8.	Crew	615	200	120
Operating Weight Empty:		$W_{OE} = 46338$		$X_{cg_{Woe}} = 680$
				$Z_{cg_{Woe}} = 160$
9.	Fuel	13878	672	127
	$W_{OE} + W_F = 60216$			$X_{cg_{Woe+Wf}} = 678$
10.	Passengers	20500	630	148
Take-off Weight		$W_{TO} = 80716$		$X_{cg_{Wto}} = 666$
				$Z_{cg_{Wto}} = 151$
	$W_{TO} - W_F = 66838$			$X_{cg_{Wto-Wf}} = 665$

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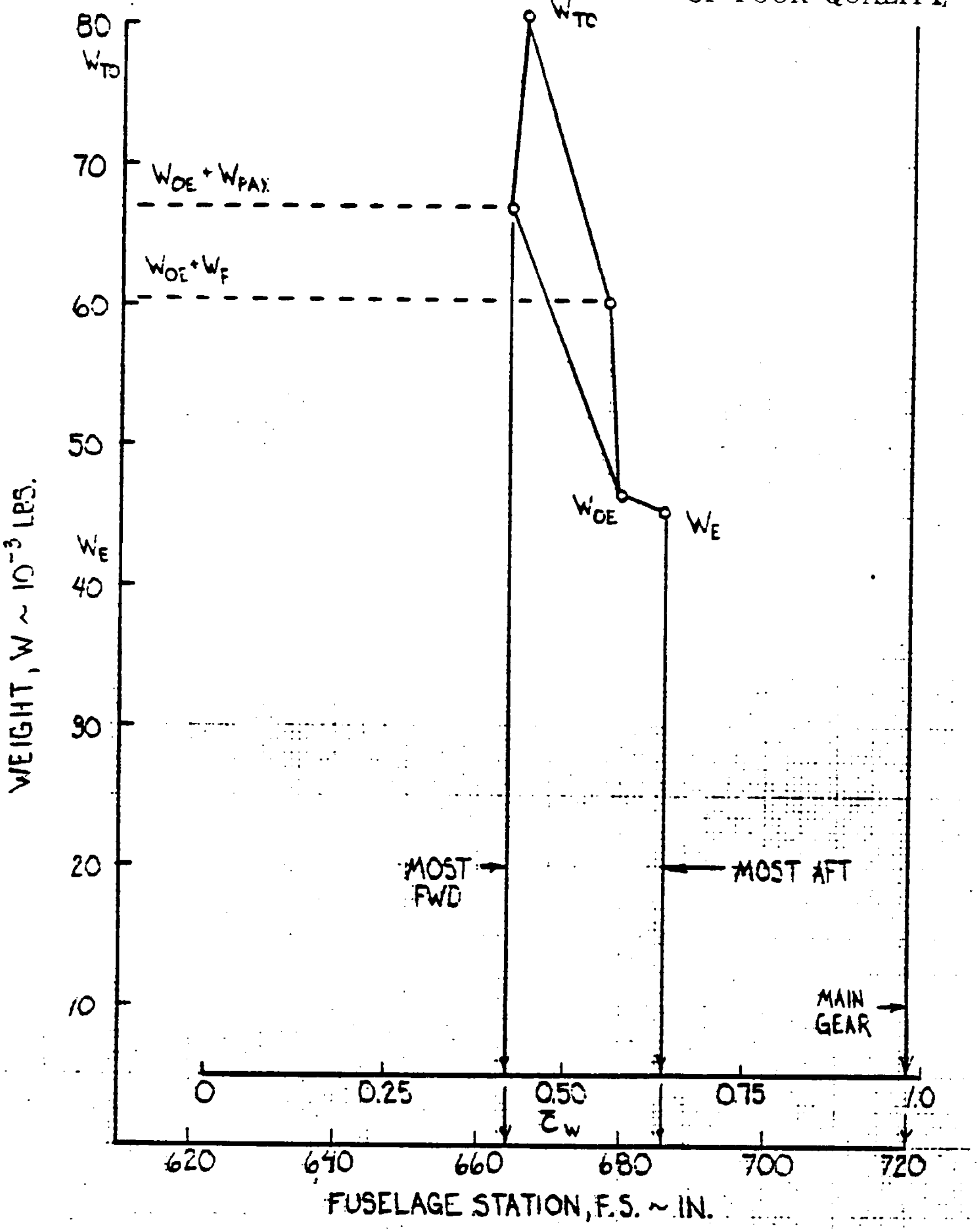

CALC	G. SWIFT	10-6	REVISED	DATE	FIGURE 2.5.3. 100-PAX TWIN BODY WEIGHT-C.G. EXCURSION DIAGRAM.	AE 790
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N.4 Class I Stability and Control Calculations

The following section provides the complete set of calculations for the Class I stability and control calculations.

100 Pax - Twin Body

$c_f = 123$

Section	Δx_i	$w_f(x_i)$	x_i	x_i/c_f
1	97	70	436.5	3.55
2	97	95	339.5	2.76
3	97	97	242.5	1.97
4	97	97	145.5	1.18
5	97	97	48.5	0.39
6	182	97	91	0.74
7	182	91	273	2.22
8	182	60	455	3.70
N	100	300	142	1.15

Note: All dimensions in inches.

The wing lift curve slope has previously been determined to be:

$$C_{L_{\alpha_w}} = 4.97 \text{ rad}^{-1} = 0.0868 \text{ deg}^{-1}$$

Thus, the $d\bar{\varepsilon}/d\alpha$ correction factor is

$$\left.\frac{d\bar{\varepsilon}}{d\alpha}\right|_{0.0868} = 1.085 \left.\frac{d\varepsilon}{d\alpha}\right|_{0.08}$$

The downwash is found from Figure 3.33 of the 550 book

Section	$d\bar{\varepsilon}/d\alpha$	$d\bar{\varepsilon}/d\alpha_{\text{corrected}}$
1	1.00	1.09
2	1.00	1.09
3	1.05	1.14
4	1.10	1.19
5	2.70	2.93

The downwash at the horizontal tail is as follows:

$l_h = 562''$

$l_h / c_r = 4.57$

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from Figure 3.24 of AE 550 book:

$1 - d\varepsilon / d\alpha = 0.70$

For sections 6, 7, 8, and N,

$$\frac{d\bar{\varepsilon}}{d\alpha} = \frac{x_i}{l_h}\left(1 - \frac{d\varepsilon}{d\alpha}\right)$$

Section	x_i	x_i/l_h	$d\bar{\varepsilon}/d\alpha$
6	91	0.162	0.113
7	273	0.486	0.340
8	455	0.810	0.567
N	142	0.253	0.177

Thus, the resulting moment is calculated from,

$$\frac{dM}{d\alpha} = \frac{\bar{q}}{36.5} \Sigma\, w_f^2(x_i)\, \Delta x_i \left.\frac{d\bar{\varepsilon}}{d\alpha}\right|_i$$

$$\frac{dM}{d\alpha} = \frac{\bar{q}}{36.5}\left[(300 + 552 + 602 + 629 + 1548 + 112 + 297 + 215) \times 2 + 922\right]$$

$$\frac{dM}{d\alpha} = 258\,\bar{q}$$

thus

$$\Delta \bar{x}_{ac_B} = -0.39 \longleftarrow$$

Step 1. Prepare a longitudinal X-plot for the airplane.

a. The c.g. leg.

The horizontal tail has been assumed to have the following characteristics:

$S_H = 130 \text{ ft}^2$ each

$W_H = 589$ lbs each

Thus, $\frac{W_H}{S_H} = 4.53$ psf

From the weight and balance analysis, the following tabulation can be made:

Table 1. Center of Gravity Location For Various Horizontal Tail Areas. (Twin Horizontal Tails)

S_H ft^2	W_H lbs	X_{cg} aft	$\bar{X}_{cg_{AFT}}$
150	680	681	0.59
200	906	684	0.62
250	1133	687	0.65
300	1359	689	0.67

b. The a.c. leg.

The following quantities must be determined:

$\bar{X}_{ac_{WB}}$, $C_{L_{\alpha_h}}$, $d\varepsilon_h/d\alpha$, $\bar{X}_{ac_h}$, $C_{L_{\alpha_{he}}}$, $d\varepsilon_{he}/d\alpha$, $\bar{X}_{ac_{he}}$

From Multhopp's integration,

$$\bar{X}_{ac_{WB}} = \bar{X}_{ac_W} + \Delta\bar{X}_{ac_B}$$

$$= 0.25 - 0.39$$

$$\bar{X}_{ac_{WB}} = -0.14 \longleftarrow$$

From the 3 view,

$$\bar{X}_{ac_h} = 6.50 \longleftarrow$$

$$\bar{X}_{ac_{he}} = 1.71 \longleftarrow$$

$$C_{L_{\alpha_H}} = \left(\frac{A}{K}\right) \frac{2\pi}{2 + \sqrt{\frac{A^2\beta^2}{\kappa^2}\left(1 + \frac{\tan^2\Lambda_{c/2}}{\beta^2}\right) + 4}} \qquad (2)$$

where,

$$K = 1 + \{(8.2 - 2.3\Lambda_{LE}) - (0.22 - 0.153\Lambda_{LE})A\}/100 \qquad (3)$$

$$\beta^2 = 1 - M_\infty^2 \qquad (4)$$

$$\kappa = a_\infty/(2\pi/\beta) \qquad (5)$$

assuming $M_\infty = 0.70$ (worst case) and $a_\infty = 2\pi$

$\beta^2 = 0.51$
$\kappa = 0.71$
$K = 1.0643$

and it is known,

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$A = 5.0$
$\Lambda_{c/2} = 28°$

therefore,

$$C_{L_{\alpha_H}} = \left(\frac{5}{1.0643}\right) \frac{2\pi}{2 + \sqrt{\frac{(25)(0.51)}{(0.5[illegible])}\left(1 + \frac{\tan^2 28°}{(0.5[illegible])}\right) + 4}}$$

$$\left.\begin{array}{l} C_{L_{\alpha_H}} = 3.45 \quad \text{at } M_\infty = 0.70 \\ C_{L_{\alpha_H}} = 3.69 \quad \text{at } M_\infty \sim 0 \end{array}\right\} \longleftarrow$$

Figures 3.25 and 3.26 of Airplane Flight Dynamics And Automatic Flight Controls, Part I, will be used to estimate $d\varepsilon/d\alpha$.

$m = 250/705 = 0.33$

$r = 610/705 = 0.87$

$\lambda = 0.60$; $A = 5$; $\Lambda_{LE} = 25°$

$d\varepsilon_h/d\alpha = 0.164$

$1 - d\varepsilon_h/d\alpha = 0.836$

The lift curve slope for the engine planform is found as follows:

$A_{h_e} = 5.56$

$\beta^2 = 0.51$

$K = 0.71$

$K_L = 1.0698$

$\Lambda_{LE} = 0°$

$$C_{L_{\alpha_{he}}} = \left(\frac{5.56}{1.0698}\right) \times \frac{2\pi}{2 + \sqrt{(5.56)^2 + 4}}$$

$$C_{L_{\alpha_{he}}} = 4.13 \text{ rad}^{-1} \longleftarrow$$

The downwash gradient is calculated as follows: (method proposed in Reference)

$$\left.\frac{d\varepsilon}{d\alpha}\right|_M = \left.\frac{d\varepsilon}{d\alpha}\right|_{M=0} \frac{C_{L_{\alpha_w}}|_M}{C_{L_{\alpha_w}}|_{M=0}} = 0.992 \left.\frac{d\varepsilon}{d\alpha}\right|_{M=0} \qquad (6)$$

where

$C_{L_{\alpha_w}}|_M = 5.16 \text{ rad}^{-1}$

$C_{L_{\alpha_w}}|_{M=0} = 5.20 \text{ rad}^{-1}$

and

$$\left.\frac{d\varepsilon}{d\alpha}\right|_{M=0} = 4.44\left(K_A K_\lambda K_H \sqrt{\cos\Lambda_{c/4}}\right)^{1.19} \qquad (7)$$

$$K_A = \frac{1}{A} - \frac{1}{1+A^{1.7}} \qquad (8)$$

$$K_\lambda = \frac{10-3\lambda}{7} \qquad (9)$$

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$$K_H = \frac{1 - h_H/b}{\sqrt[3]{\frac{2 l_H}{b}}} \qquad (10)$$

where

	$A_w = 15$	$\lambda_w = 0.40$
$l_{h_e} = 171''$	$h_{h_e} = 90''$	$b_w = 118'$

$K_A = 0.0568$
$K_\lambda = 1.257$
$K_H = 1.504$

$\left.\frac{d\varepsilon}{d\alpha}\right|_{M=0} = 0.306$ and $\left.\frac{d\varepsilon}{d\alpha}\right|_{M} = 0.303$

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and $(1 - d\varepsilon/d\alpha)_{ne} = 0.697$

The a.c. leg is calculated by (Eqn. 11)

$$\bar{X}_{ac_A} = \frac{\left[\bar{X}_{ac_{WB}} + \left\{C_{L_{\alpha h}}\left(1-\frac{d\varepsilon_h}{d\alpha}\right)\left(\frac{S_h}{S}\right)\bar{X}_{ac_h}\right\}/C_{L_{\alpha WB}} + \left\{C_{L_{\alpha ne}}\left(1-\frac{d\varepsilon_{ne}}{d\alpha}\right)\left(\frac{S_{ne}}{S}\right)\bar{X}_{ac_{ne}}\right\}/C_{L_{\alpha WB}}\right]}{\left[1 + \left\{C_{L_{\alpha h}}\left(1-\frac{d\varepsilon}{d\alpha}\right)_h\left(\frac{S_h}{S}\right)/C_{L_{\alpha WB}}\right\} + \left\{C_{L_{\alpha ne}}\left(1-\frac{d\varepsilon}{d\alpha}\right)_{ne}\left(\frac{S_{ne}}{S}\right)\right\}/C_{L_{\alpha WB}}\right]}$$

In the stability and control analysis for the 50 passenge single body, it was found that

$S_h = 102\ ft^2$

thus, it will be assumed for commonality that the horizontal tails on the 100-passenger twin body are the same as that on the 50 passenger. Thus,

$S_h = 204\ ft^2$

Thus, the above equation reduces to

$$\bar{X}_{ac_A} = \frac{[-0.14 + 0.803 + 0.954(S_{ne}/S)]}{[1 + 0.1235 + 0.558(S_{ne}/S)]} \qquad M = 0.70$$

Table 2. Aerodynamic Center Location For Various Engine Planform Support Areas.

S_{He} ft^2	$\frac{S_{ne}}{S}$	$\bar{X}_{ac_A}$
0	0	0.590
100	0.108	0.647
200	0.217	0.699
300	0.325	0.746

By assuming the horizontal tail weight is fixed at

$$W_h = 924 \text{ lbs} \longrightarrow \bar{X}_{cg_h} = 0.620$$

Then for the airplane to be inherently stable with a 5 percent static margin,

$$\bar{X}_{ac_E} = 0.670$$

and

$$S_{h_E} = 150 \text{ ft}^2 \longleftarrow$$

This corresponds to $W_{H_E} = 680$ lbs which has been assumed to be included in the engine weight. Class II weight estimates may reveal that this assumption be re-evaluated.

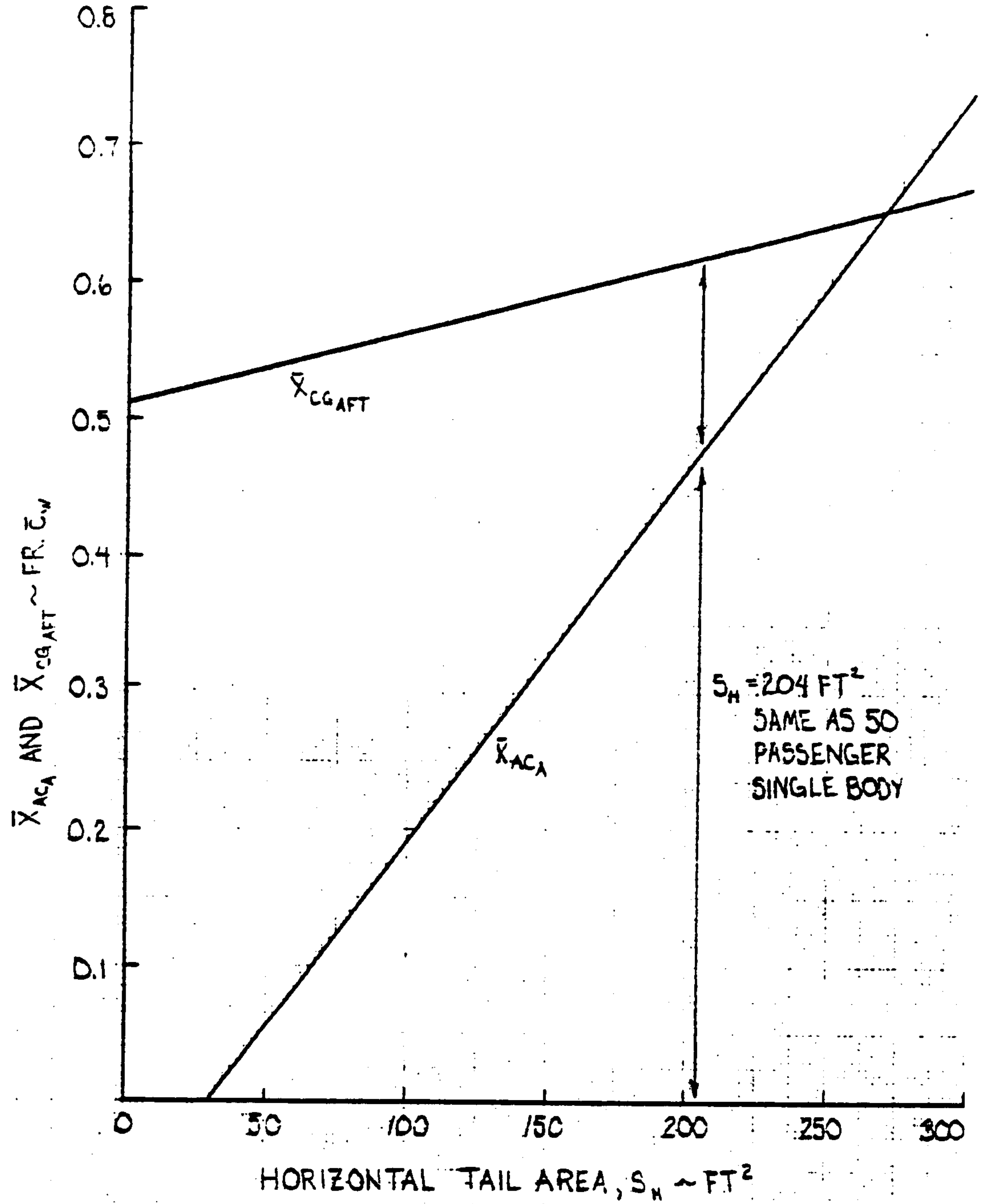

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FIGURE 2.5.4. LONGITUDINAL X-PLOT FOR THE 100-PAX TWIN BODY DESIGN.

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Step 4. Prepare a directional X-plot for the airplane.

Table 1 provides the c.g. data.

The C_{n_β} leg of the X-plot follows from:

$$C_{n_\beta} = C_{n_{\beta_{WB}}} + C_{L_{\alpha_V}} (S_V/S)(x_V/b) \qquad (7)$$

From the preliminary engine out computations of 9/25/86,

$C_{L_{\alpha_V}} = 2.14 \text{ rad}^{-1}$ $S = 923 \text{ ft}^2$

$X_{V_s} = 41.3$ ft $b = 118$ ft

From Methods For Estimating Stability And Control Derivatives of Conventional Subsonic Airplanes,

$$C_{n_{\beta_{WB}}} = C_{n_{\beta_W}} + C_{n_{\beta_B}} \qquad (8)$$

$C_{n_{\beta_W}} \sim 0$: The wing contribution is very small except at high angles of attack.

$$C_{n_{\beta_B}} = -57.3\ K_N K_{R_l} \frac{2 S_{B_s}}{S} \frac{l_B}{b} \quad (\text{rad}^{-1}) \qquad (9)$$

where,

S_{B_s} = Side body area = 624 ft^2

$l_b = 94.6$ ft

$x_m = 49.1$ ft

$x_m / l_b = 0.52$

$l_b^2/S_{B_s} = 14.3$

$h_1 = 98$ in.

$h_2 = 89$ in.

$(h_1/h_2)^{1/2} = 1.05$

$w = 98$ in $h/w = 1.0$

$K_N = 0.001$

$$R_{l_f} = \frac{\rho V l_f}{\mu} \qquad (10)$$

thus,

$R_{l_f} = 152 \times 10^6$ at take-off ($V \doteq 150$ knots)

$R_{l_f} = 187 \times 10^6$ at cruise ($M = 0.7$ at 35000 ft)

Since cruise R_{l_f} is larger

$K_{R_l} = 2.08$

$Cn_{\beta_B} = -(57.3)(0.001)(2.08)(634/923)(94.6/118) \times 2$

$Cn_{\beta_B} = -0.129 \text{ rad}^{-1}$

and

$Cn_{\beta} = -0.129 + 0.749(S_v/S)$

Table 3. Directional Stability For Various Tail Areas. (Twin Vertical Tails)

S_v ft^2	S_v/S	Cn_β rad^{-1}
100	0.108	-0.048
200	0.217	0.034
300	0.325	0.114

Figure 2.5.5 is the related graph.

Step 5. Determine whether or not the airplane being designed needs to have 'inherent' or 'de factor' directional stability.

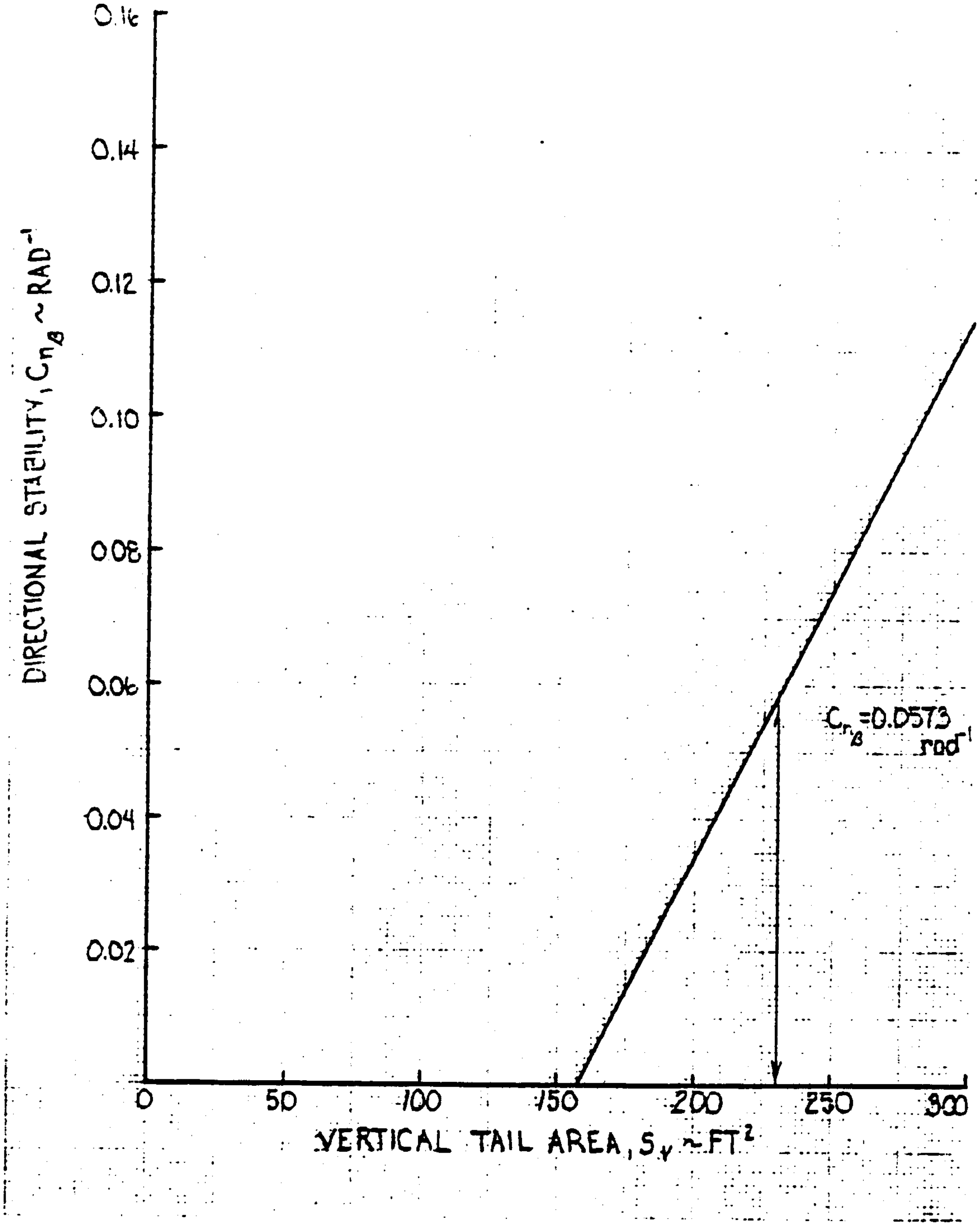
This airplane will be inherently stable.

Step 6. Assume that the overall level of directional stability must be,

$Cn_\beta = 0.0573 \text{ rad}^{-1}$

From Figure , this corresponds to

$S_v = 230 \text{ ft}^2$



CALC	G. SWIFT	10-8	REVISED	DATE	FIGURE 2.5.5. 100-PAX TWIN BODY DIRECTIONAL X-PLOT.	AE 790
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MIN. FOR CONTROL SPEED WITH ONE ENGINE INOPERATIVE

Step 7. Determine the critical engine-out yawing moment from:

$$N_{t_{crit}} = T_{TO_e} y_t \quad (11)$$

where from calculations done on 9/25/86,

$T_{TO_e} = 24{,}700$ lbs

$y_t = 7$ ft. (maximum)

$N_{t_{crit}} = 172{,}700$ ftlbs.

Step 8. Determine the value of drag induced yawing moment due to the inoperative engine from:

$$N_D = 0.4\, N_{t_{crit}} \quad (12)$$

$N_D = 69{,}100$ ftlbs

Step 9. Calculate the maximum allowable Vmc from:

$$V_{mc} = 1.2\, V_S$$

where

$V_S = 172$ fps

$V_{mc} = 206$ fps

Step 10. Calculate the rudder deflection required to hold the engine out condition at Vmc from:

$$\delta_r = (N_D + N_{t_{crit}}) / \bar{q}_{mc}\, S\, b\, C_{n_{\delta_R}} \quad (13)$$

where

$\bar{q}_{mc} = 50.4$ psf

$S = 923$ ft^2

$b = 118$ ft

$$\delta_R = \frac{0.044}{C_{n_{\delta_R}}}$$

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Cn_{δ_R} may be computed from

$$Cn_{\delta_R} = -Cy_{\delta_R} \left(\frac{l_v \cos\alpha + Z_v \sin\alpha}{b} \right) \qquad (14)$$

where

$l_v = 40.5$ ft.

$Z_v =$ N.A.

$b = 118$ ft.

$\alpha = 0°$ (assumed to be small)

thus,

$$Cn_{\delta_R} = -Cy_{\delta_R}(0.343)$$

Cy_{δ_R} can be calculated from

$$Cy_{\delta_R} = C_{L_{\alpha_v}} \left[\frac{(\alpha_\delta)_{C_L}}{(\alpha_\delta)_{C_l}} \right] (\alpha_\delta)_{C_l} K' K_b \frac{S_v}{S} \qquad (15)$$

where

$C_{L_{\alpha_v}} = 2.14 \text{ rad}^{-1}$

$\frac{(\alpha_\delta)_{C_L}}{(\alpha_\delta)_{C_l}} = 1.14$

$(\alpha_\delta)_{C_l} = -0.7$

$K' = 0.65$

$K_b = 1.0$

thus

$$Cy_{\delta_R} = -0.0012(S_v)$$

and

$$Cn_{\delta_R} = 0.0004\, S_v$$

where

$C_{n_{\delta}} = 0.0004(S_v)$

and

$\delta = \frac{107}{S_v \cdot 2}$ (rad)

Table 4. Vertical Tail Area Required For OEI

δ_R deg	S_v ft²
25	122
30	102
35	88

Thus, the maximum vertical tail area required is,

$S_v = 122$ ft² $\longleftarrow$

N.5 Class 1 Drag Polar Calculations

This section provides the calculations for the 100 passenger twin body Class 1 drag polars.

Step 1. List all airplane components which contribute to wetted area, compute the wetted area of these components. Find the sum, S_{WET}.

The components that contribute to wetted area are:

1. Fuselage
2. Wing
3. Empennage
4. Nacelles
5. Pylons

1. Wetted Area for Planforms.

The wetted area of the planform can be found from:

$$S_{wet_{plf}} = 2\, S_{exp_{plf}} \{ 1 + 0.25\,(t/c)_r (1 + \tau\lambda)/(1+\lambda) \}$$

where,

$$S_{exp_{plf}} = \frac{b_{exp}}{2}\, C_{r_{exp}} (1+\lambda)$$

$$= (38.1/2)(9.17)(1.45) + (10 \times 25)$$

$$= 503.1 \text{ ft}^2$$

$(t/c)_r = 0.13$; $\tau = 1.3$; $\lambda = 0.4$

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thus

$S_{wet_{plf}} = 1042 \text{ ft}^2$

For the horizontal tail,

$S_{exp} = 102 \text{ ft}^2 + 201 \text{ ft}^2$

$(t/c)_r = 0.12$; $\tau = 1.2$; $\lambda = 0.5$

$S_{wet_{hor}} = 625 \text{ ft}^2$

For the vertical tail,

$S_{exp} = 280 \text{ ft}^2$

$(t/c)_r = 0.13$; $\tau = 1.08$; $\lambda = 0.50$

$S_{wet_{ver}} = 567 \text{ ft}^2$

2. Wetted Area for Fuselages.

For fuselages with cylindrical mid-sections:

$$S_{wet_{fus}} = \pi D_f l_f (1 - 2/\lambda_f)^{2/3} (1 + 1/\lambda_f^2)$$

where

$\lambda_f = 11.7$, the fuselage fineness ratio

$$S_{wet_{fus}} = \pi (8.08)(94.6)(1 - 2/11.7)^{2/3}(1 + 1/11.7^2)$$

$$S_{wet_{fus}} = 2 \times S_{wet_{fus}} \quad \text{(twin body)}$$

$$S_{wet_{fus}} = 4270 \text{ ft}^2$$

3. Wetted Area for Nacelles.

The nacelle area will be estimated by.

$$S_{wet_{nac}} = \pi \, l_{eng} \, d_{eng}$$

$$= \pi (12.5)(5.0) \times 2$$

$$S_{wet_{nac}} = 393 \text{ ft}^2$$

4. Pylon Wetted Area.

$$S_{wet} = 2 S_{exp_{plf}} [1 + .25 (t/c)_r (1 + \tau\lambda)/(1 + \lambda)]$$

where,

$\lambda = 1$; $(t/c)_r = 0.12$; $\tau = 1.0$

$S_{exp} = 153 \text{ ft}^2$

thus

$$S_{wet} = 315 \text{ ft}^2$$

5. Summary of Wetted Areas.

Component	Wetted Area ft^2
Wing	1042
Horizontal Tail	625
Vertical Tails	567
Fuselages	4270
Engine Nacelles	393
Engine Pylons	315
TOTAL	7212 ft^2

Step 2. Using figures 3.21 of Part I find the equivalent parasite area, 'f' of the airplane.

$$f = 17.0\ ft^2 \quad (C_f = 0.0025)$$

Step 3. Determine the 'clean' zero lift drag coefficient:

$$C_{D_0} = 17.0/923 = 0.0184$$

Step 4. Find the compressibility drag increment of the airplane from Figure 12.7.

$\Delta C_{D_0} = 0.0002$ for compressibility

Step 5. The following drag increments will be assumed.

Configuration	ΔC_{D_0}
Takeoff gear	0.015
Landing gear	0.015
Landing flaps	0.075

Step 6. Determine the drag polars.

Take-off: $C_D = 0.0334 + 0.0265\, C_L^2$

$C_{D_0} = 0.0334$ $(L/D)_{max} = 16.8$
$e = 0.80$
$A = 15$

Cruise: $C_D = 0.0186 + 0.0250\, C_L^2$

$C_{D_0} = 0.0186$ $(L/D)_{max} = 23.2$
$e = 0.85$
$A = 15$

Landing: $C_D = 0.1084 + 0.0265\, C_L^2$

$C_{D_0} = 0.1084$ $(L/D)_{max} = 9.32$
$e = 0.8$
$A = 15$

Step 7. Determine Cruise L/D.

Assuming $C_{L_{cruise}} = 0.3$

$(L/D)_{cr} = 14.4$ ⟵

By taking a 10% reduction in C_{D_0}

$(L/D)_{cr} = 15.8$

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N.6 Class I Inertia Calculations

The following provides the Class I inertia calculations for the 100 passenger twin body design.

Step 1. Evaluate I_{xx}.

I_{xx} will be calculated later after completion of Class II weight and balance. Class I methods are not accurate enough for the twin body configuration.

Step 2. Evaluate I_{yy} and I_{zz}.

$W_{TO} = 80{,}516$ lbs

$W_{OE} = 46{,}138$ lbs

$$\left.\begin{aligned} b &= 118 \text{ ft} \\ L &= 104 \text{ ft} \end{aligned}\right\} e = 111 \text{ ft}$$

The non-dimensional radius for this airplane about the y-axis is assumed to be

$\bar{R}_y = 0.4$ (Table B7a, Part V)

and about the z-axis,

$\bar{R}_z = 0.5$

The inertias can be calculated from

$$I_{yy} = L^2 W (\bar{R}_y)^2 / 4g \qquad (1)$$

$$I_{zz} = (e^2 W (\bar{R}_z)^2 / 4g + md^2) \times 2 \qquad (2)$$

Equation (2) assumes that $\bar{R}_z$ applies to each body. Thus, the inertia is calculated for each body and then translated to the airplane c.g.. ($d = 16.7$ ft and $W_{OE} = 23069$ lbs, $W_{TO} = 40{,}258$ lbs for each body)

At W_{TO}:

$I_{yy} = 1.082 \times 10^6$ slug ft^2

$I_{zz} = 2.623 \times 10^6$ slug ft^2

At W_{OE}:

$I_{yy} = 6.199 \times 10^5$ slug ft^2

$I_{zz} = 1.503 \times 10^6$ slug ft^2

Step 3. Compare inertias with known data.

The figures on the following pages show how the 100 passenger twin body compares with existing data.

I_{yy} may be a little high; however, the engine pylons may validate the higher I_{yy} assumed.

I_{zz} appears to be double that of existing conventional configurations - for a good reason. The twin body design validates the I_{zz} values assumed.

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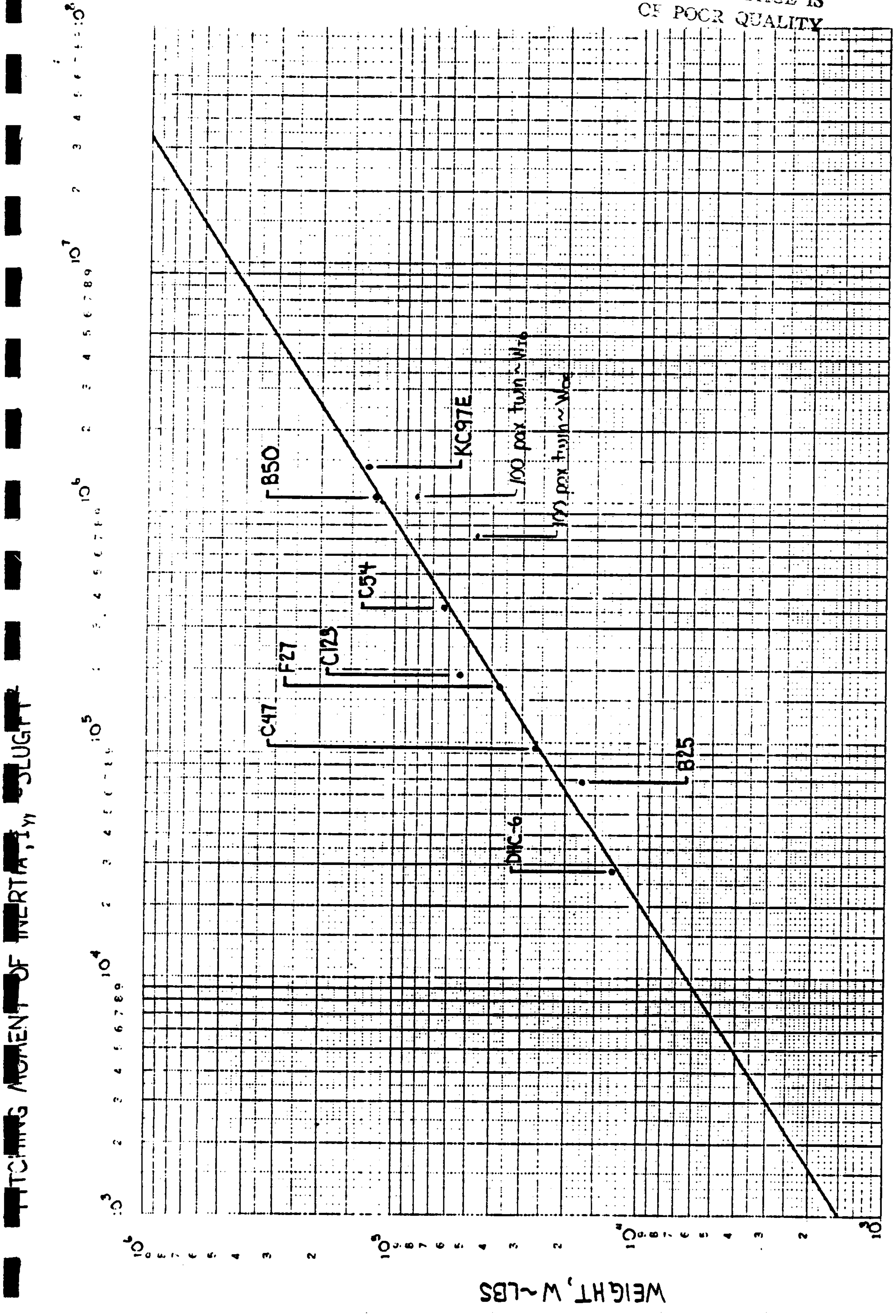

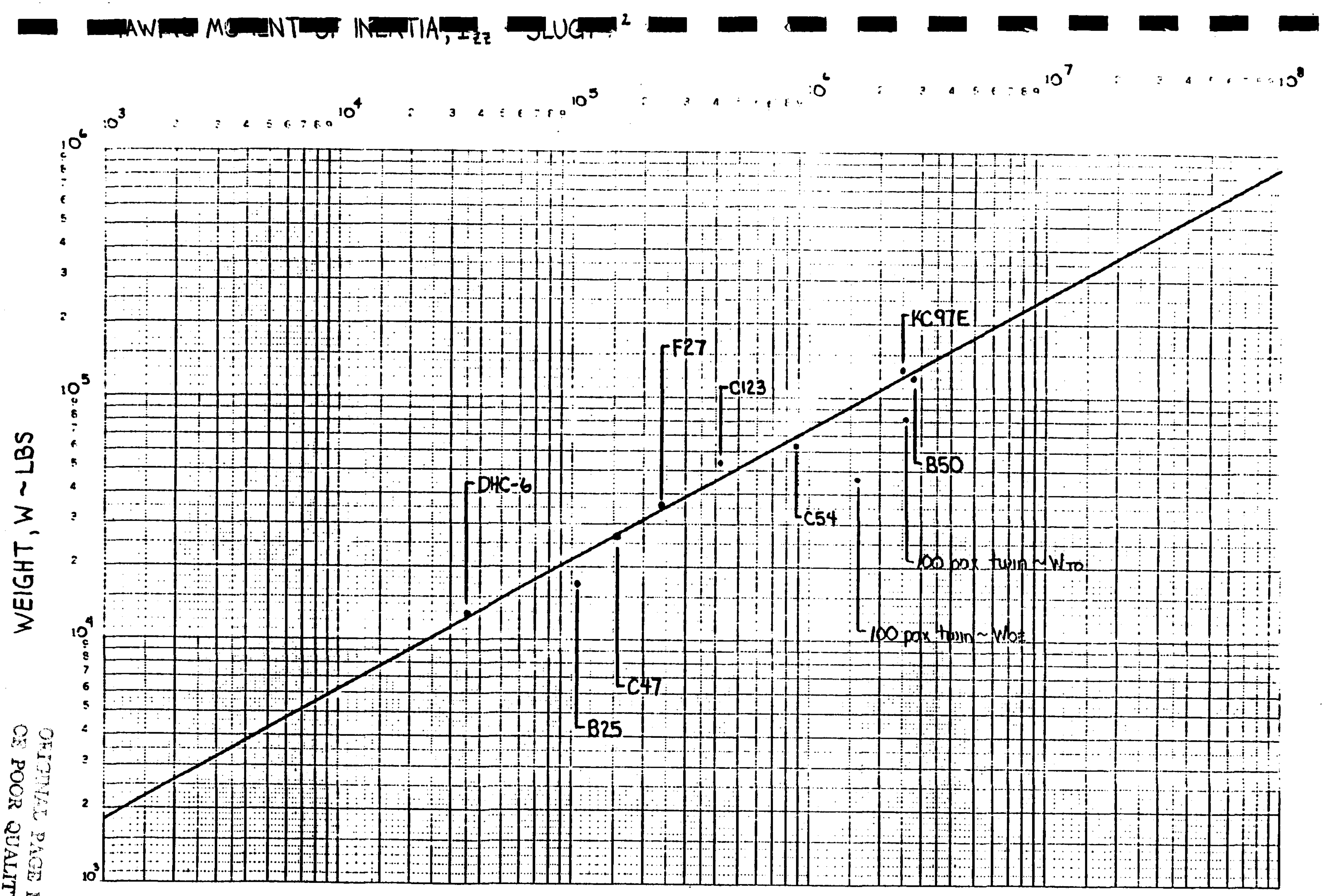
YAWING MOMENT OF INERTIA, I_{zz} ~ SLUG FT^2
10^3
10^4
10^5
10^6
10^7
10^8
WEIGHT, W ~ LBS
10^3
10^4
10^5
10^6
DHC-6
F27
C123
KC97E
B50
C54
100 pax twin ~ W_{TO}
100 pax twin ~ W_{OE}
C47
B25

APPENDIX O

PRELIMINARY DESIGN
WING STRUCTURAL WEIGHT CALCULATIONS
FOR
50 PASSENGER AND 100 PASSENGER
COMMUTER AIRPLANES

TABLE OF CONTENTS

1. WING WEIGHT CALCULATIONS

DETERMINATION OF STRUCTURAL WING WEIGHT FOR 50 PASSENGER COMMUTER

From section 5.1.2.1, GD (General Dynamics) methodology was used in determining the wing weight estimations for commerical transport airplanes. The following equation will illustrate this methodology.

$$W_w = \frac{\left\{ 0.00428 \, (S)^{0.48} (A) (M_H)^{0.43} (W_{TO} \, n_{ULT})^{0.84} (\lambda)^{0.14} \right\}}{\left[(100 \, (t/c)_{MAX})^{0.76} \, (\cos \Lambda_{\frac{1}{2}\tau})^{1.54} \right]}$$

Note: This equation is only valid for the following parameters ranges.

$M = 0.4 - 0.8$

$(t/c)_{MAX} = 0.08 - 0.15$

$A = 4 - 12$

Through reseach of airplanes with similar performance requirements the following assumptions were made for the 50 passenger airplane.

$W_{TO} = 45{,}000$ lbs

$S = 600$ ft

$\lambda = 0.3$

The design limit load factor, n , was determined from equation 4.13 of Reference 1, which is as follows:

$$n_{LIM} \geq 2.1 + \{24{,}000/(W_{TO} + 10{,}000)\}$$

Exceptions

n_{LIM} need not be greater than 3.8

$n_{LIM} = 4.4$ for utility airplanes

$n_{LIM} = 6.0$ for acrobatic airplanes

where:

$n_{ULT} = 1.5(n_{LIM})$

2. RESULTS OF WING WEIGHT CALCULATIONS

For the 50 passenger commuter

$n_{ULT} = 3.80$

For the 100 passenger commuter ($W_{TO} = 110{,}000$ lbs)

$n_{ULT} = 3.45$

However, One should keep in mind the influence of the negative ultimate load factor, n_{ULT}. In the weight estimations this value isn't critical but from a structural analysis (the critical mode of failure) view this factor can be the dominating driver.

TABLE 2.1 50 PASSENGER COMMUTER

W_{TO} = 45,000, S = 600 ft^2, λ = 0.3, $(t/c)_{MAX}$ = 0.15

ASPECT RATIO \ SWEEP ($\Lambda_{c/4}$) WEIGHT X10^3	5	10	15	20	25	30
8	1.81	1.87	1.89	1.98	2.09	2.25
9	2.04	2.07	2.14	2.23	2.36	2.53
10	2.26	2.30	2.37	2.47	2.62	2.81
11	2.49	2.53	2.61	2.72	2.88	3.09
12	2.71	2.76	2.85	2.97	3.14	3.37
13	2.94	2.99	3.08	3.22	3.40	3.65
14	3.17	3.22	3.32	3.46	3.66	3.93

TABLE 2.2 100 PASSENGER COMMUTER

W_{TO} = 110,000 lbs, S = 1000 ft^2, λ = 0.3, $(t/c)_{MAX}$ = 0.15

ASPECT RATIO \ SWEEP ($\Lambda_{c/4}$) WEIGHT X10^3	5	10	15	20	25	30
8	4.52	4.60	4.74	4.94	5.23	5.61
9	5.08	5.17	5.33	5.56	5.88	6.31
10	5.65	5.75	5.92	6.18	6.53	7.01
11	6.21	6.32	6.51	6.80	7.19	7.71
12	6.78	6.90	7.11	7.41	7.84	8.41
13	7.34	7.47	7.70	8.03	8.49	9.11
14	7.91	8.05	8.29	8.65	9.15	9.81

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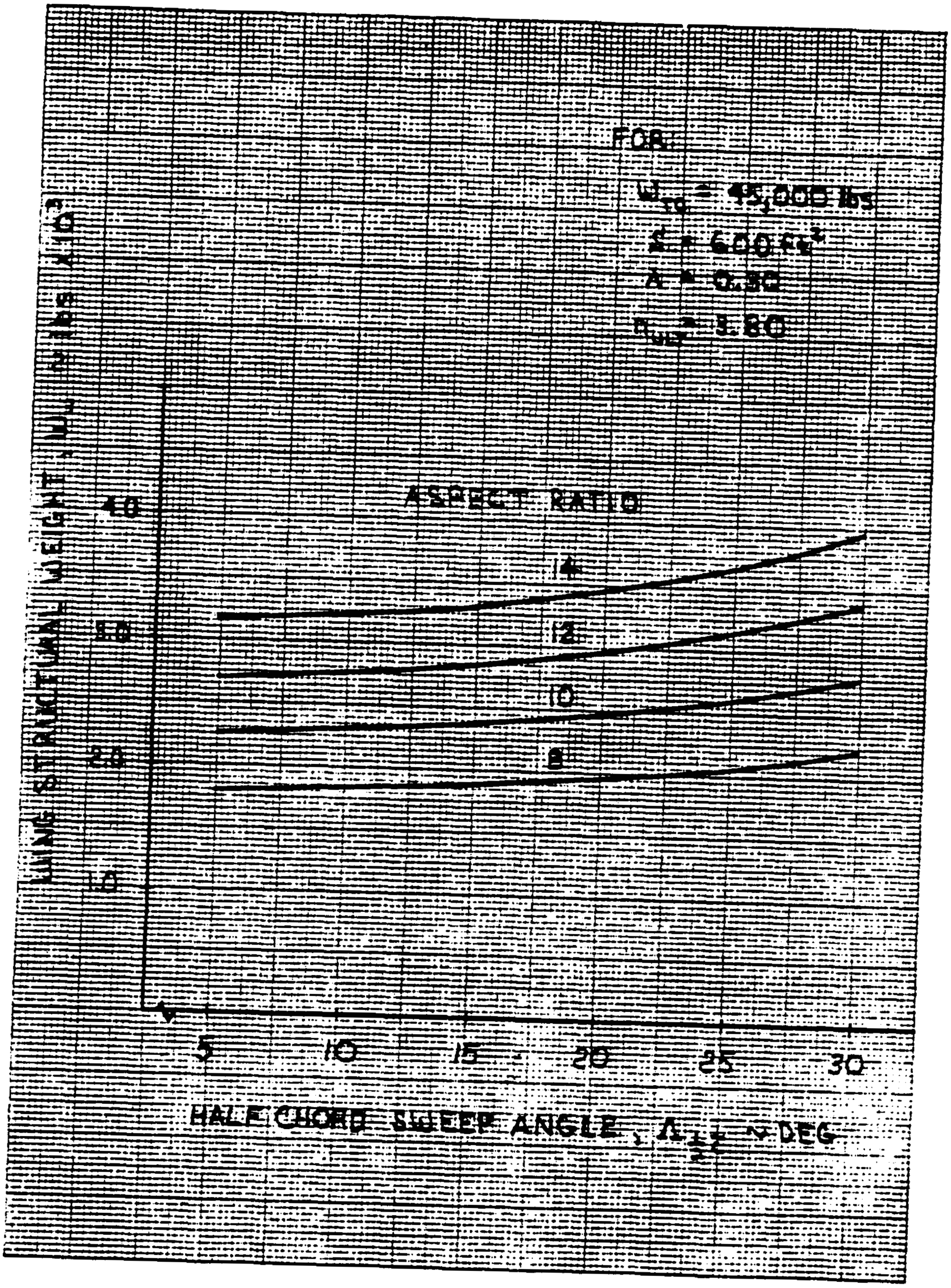

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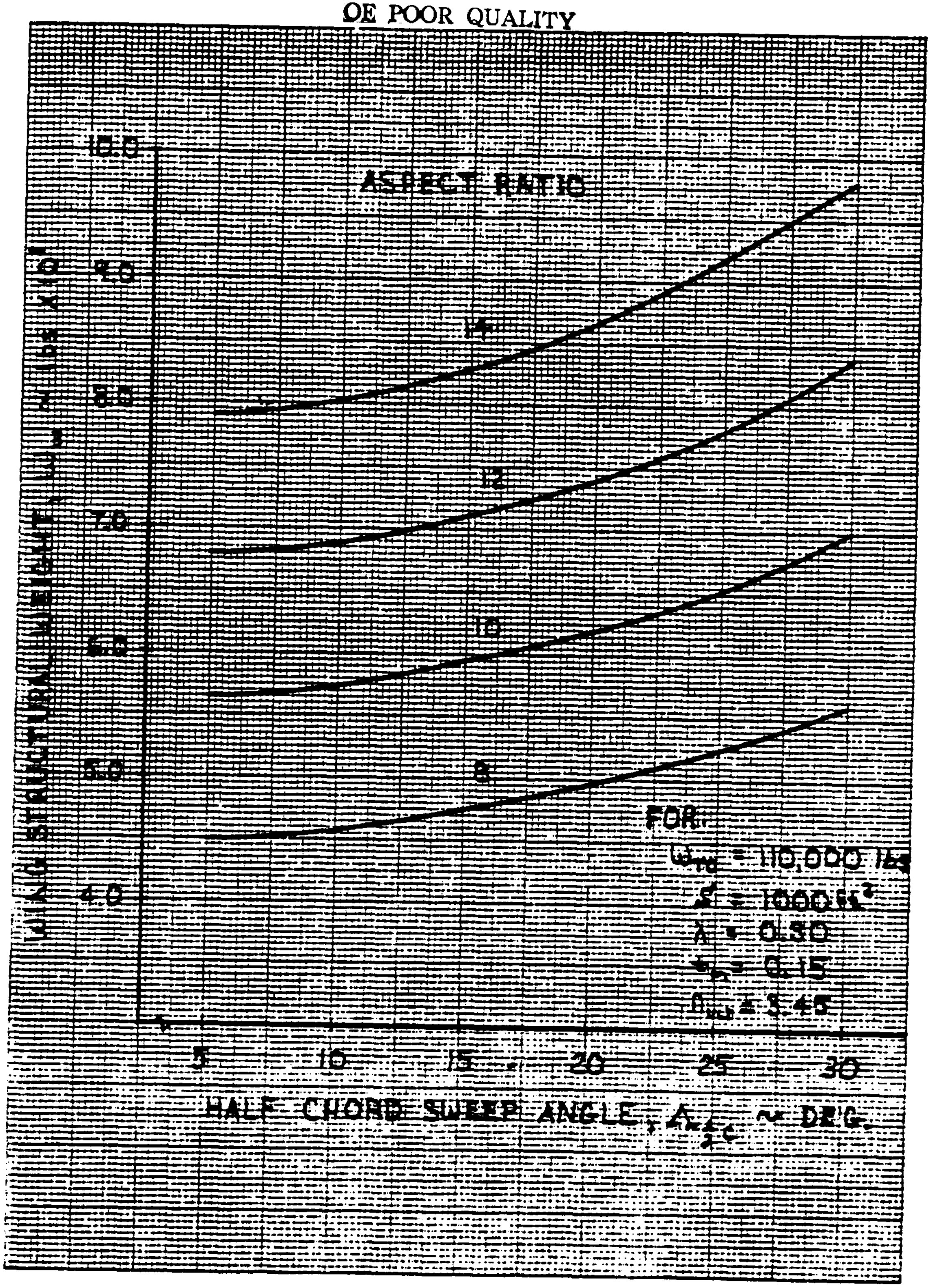

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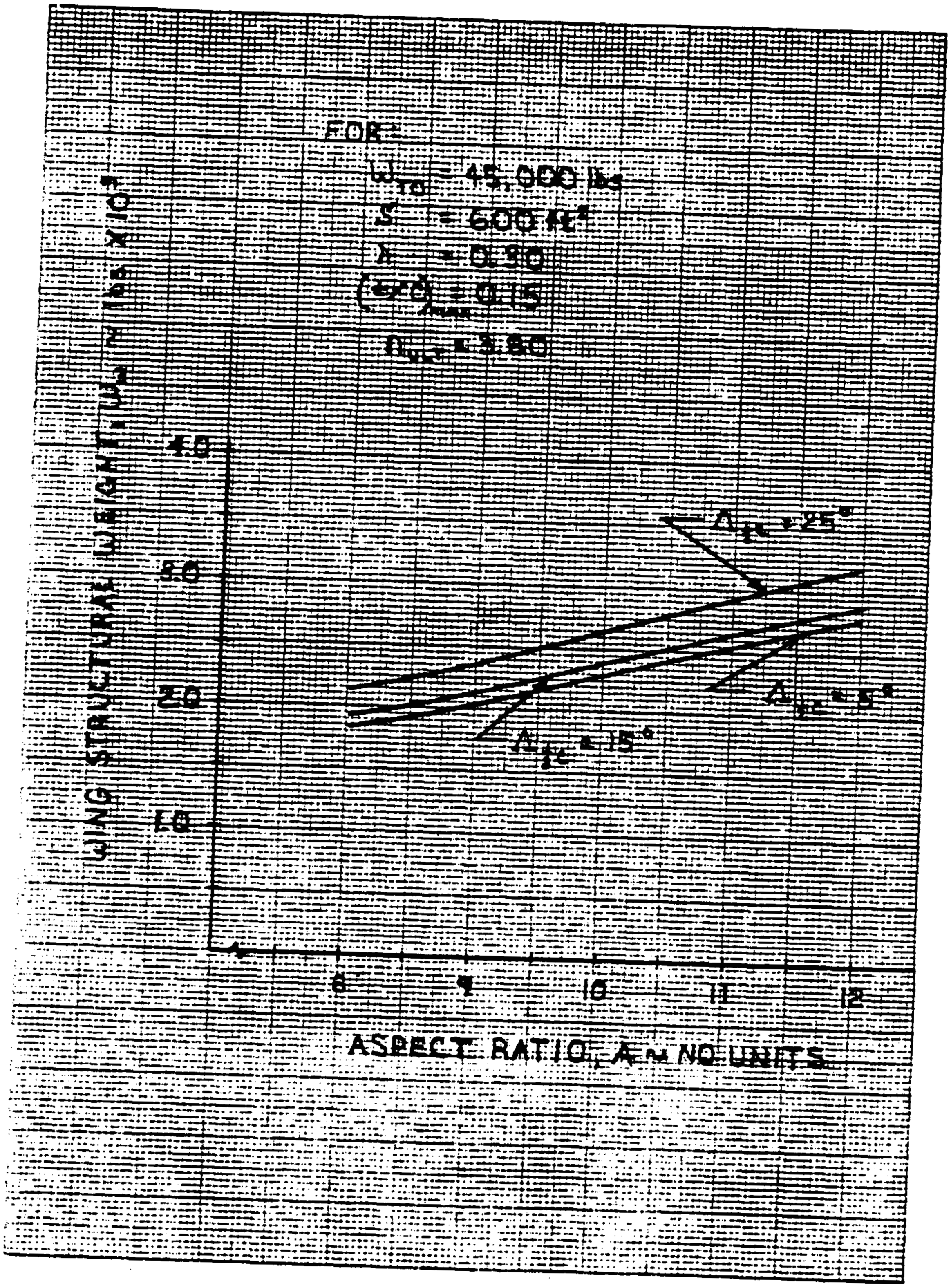

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EFFECT OF ASPECT RATIO ON WING STRUCTURAL WEIGHT

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FIGURE 03

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